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ALTITUDE TESTING OF THE J-2 ROCKET ENGINE IN PROPULSION ENGINE TEST CELL (J-4) (TESTS J4-1554-12 THROUGH J4-1554-19)

J. N. Simpson, F. D. Cantrell, and C. H. Kunz

ARO, Inc.

September 1967

LARGE ROCKET FACILITY
ARNOLD ENGINEERING DEVELOPMENT CENTER
AIR FORCE SYSTEMS COMMAND
ARNOLD AIR FORCE STATION, TENNESSEE

NOTICES

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Per AF Letter
dtd 12 July 74
Signed William O. Cole.

FOREWORD

The work reported herein was sponsored by the National Aeronautics and Space Administration, Marshall Space Flight Center (NASA/MSFC) under System 921E, Project 9194.

The results of the tests presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under Contract AF40(600)-1200. Program direction was provided by NASA/MSFC; engineering liaison was provided by the J-2 engine manufacturer, Rocketdyne Division of North American Aviation, Inc., and by the manufacturer of the S-IVB stage, Douglas Aircraft Company. Testing was conducted during the period from December 2, 1966, to February 5, 1967, in Propulsion Engine Test Cell (J-4) of the Large Rocket Facility (LRF) under ARO Project No. KA1554. The manuscript was submitted for publication on May 15, 1967.

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This technical report has been reviewed and is approved.

Harold Nelson, Jr.
Captain, USAF
AF Representative, LRF
Directorate of Test

Leonard T. Glaser
Colonel, USAF
Director of Test

ABSTRACT

Eight test periods involving a total of 14 starts of the J-2 rocket engine were conducted at pressure altitudes ranging from 93,000 to 111,000 ft. These tests were a continuation of an environmental verification and start transient investigation on a flight configuration J-2 engine (S/N J-2052). Firing durations ranged up to 50 sec; a total of 293.6 sec of engine operating time was accumulated during the test period. Unexpected excessive gas generator temperatures experienced on restart tests under simulated orbital restart conditions necessitated reorientation of the objectives for this test series. Satisfactory engine restart was subsequently obtained at turbine crossover duct conditions predicted for Saturn V (flight 501) restart with the propellant utilization valve in the full open position.

per letter
dtd 12 July
signed William
E. Cole

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II. TABLES

NOMENCLATURE

A	Area, in.^{2}
ASI	Augmented spark igniter
C*	Characteristic velocity, ft/sec
ES	Engine start, time at which He control and ignition phase solenoids are energized
ESCS	Engine safety cutoff system
GG	Gas generator
GGOT	Gas generator outlet temperature
MOV	Main oxidizer valve
O/F	Oxidizer-to-fuel weight flow ratio
OTBV	Oxidizer turbine bypass valve
PU	Propellant utilization
RTT	Resistance temperature transducer
STDV	Start tank discharge valve
T_o	Time at which the opening signal is applied to the STDV solenoid
VSC	Vibration safety count, an indicator of engine vibration in excess of 150 g

SUBSCRIPTS

e	exit
f	force
m	mass
t	throat

SECTION I
INTRODUCTION

In a continuing test program at the Arnold Engineering Development Center (AEDC), the J-2 engine (S/N J-2052) was subjected to various simulated space conditions to fulfill testing requirements as specified by the National Aeronautics and Space Administration, Marshall Space Flight Center (NASA/MSFC). Testing of the J-2 engine and full-scale S-IVB static test stage has been in progress since July, 1966, in Propulsion Engine Test Cell (J-4) of the Large Rocket Facility (LRF). The initial tests of this program are reported in Ref. 1. The tests reported herein, J4-1554-12 through J4-1554-19, were conducted between December 2, 1966, and February 5, 1967. During this period, 14 engine starts (ES) at pressure altitudes ranging from 93,000 to 111,000 ft were obtained, yielding a total engine operating time of 293.6 sec.

The initial objective of this test series was to verify that J-2 engine performance (at environmental conditions to which it will be subjected in flight) was as previously determined in lower altitude tests. Both first burn and restart mission simulations were to be obtained for the S-IVB stage (applicable to the Saturn IB and Saturn V vehicles) and the S-II stage (applicable to the Saturn V vehicle). However, the results of the initial tests in this series caused NASA/MSFC to revise the immediate program objectives. The revised objectives were to study combined altitude and environmental effects on the ES cycle, with particular attention focused on the engine restart application. A launch restriction on the first Saturn V (flight 501) was imposed by NASA until successful orbital restart could be proved at AEDC. To prevent a launch schedule impact, NASA established an accelerated test schedule at AEDC (for the accelerated schedule, one test period, involving four engine firings, was planned each week for ten consecutive weeks beginning January 9, 1967).

SECTION II
APPARATUS

2.1 TEST ARTICLE

The test article was a J-2 rocket engine (S/N J-2052), designed and developed by Rocketdyne Division of North American Aviation, Inc., and used in connection with a S-IVB battleship stage, designed and developed by Douglas Aircraft Company. The fluid dynamic characteristics of the battleship stage are identical to the S-IVB flight vehicle. The J-2 rocket engine is a multiple-restart engine that utilizes liquid oxygen (LO_2) and liquid hydrogen (LH_2) as propellants and is designed to be used singularly or in clusters. Thrust rating of the engine is 225,000 lb_f at an oxidizer-to-fuel mixture ratio (O/F) of 5.5. A cutaway view of the flight version of the test article is presented in Fig. 1. Nominal engine performance at rated conditions is presented in Table I.

The major engine components at the beginning of this test period are shown in Table II. All engine configuration changes accomplished during this test period are presented in Table III. Before the first test in this series, the thrust chamber was insulated with Larodyne®, a standard, pre-cast, silicone elastomer insulation for an S-IVB configuration engine (Fig. 2). The insulation was applied to the thrust chamber between the fuel inlet manifold and the nozzle exit. The engine remained in this configuration through test 18. Subsequently, the insulation was removed, and heater blankets were installed. The heater blankets (P/N 105906 through 105906-15) consisted of 16 sections of film-type, electrical heating elements applied to the thrust chamber between the throat and nozzle exit; the blankets were covered with aluminum foil secured with an overlay of wire mesh. The blankets are planned for thrust chamber heating to more accurately simulate orbital restart conditions; these blankets were not utilized for heating in this test series.

2.1.1 J-2 Rocket Engine

The J-2 rocket engine (Ref. 2, Fig. 3) features the following major components:

1. A regenerative fuel-cooled, tubular-wall, bell-shaped thrust chamber (Fig. 4) with a throat area (A_t) of 170.4 in.^{2} and an expansion ratio (A_e/A_t) of 27.1. Thrust chamber length (from the injector flange to nozzle exit) is 107 in.
2. A concentric-orificed, porous-faced thrust chamber injector (Fig. 5). Orifice areas for fuel and oxidizer injection are 25 and 16 in.^{2}, respectively. Fuel flow through the porous face of the injector is from 3 to 4 percent of thrust chamber fuel flow rate.
3. An augmented spark igniter (ASI) assembly (Fig. 6) to which fuel and oxidizer are routed and ignited at ES to provide ignition energy for main chamber propellants.

4. A fuel turbopump which is composed of a two-stage turbine-stator assembly - an inducer and a seven-stage, axial-flow pump, rotor-stator assembly. The pump is self lubricated and nominally produces a head rise of 35,517 ft of hydrogen (H_2) at a flow rate of 8414 gpm for a rotor speed of 26,702 rpm at rated conditions.

5. An oxidizer turbopump which is composed of a two-stage turbine-stator assembly and an inducer and single-stage centrifugal pump. The pump is self lubricated and nominally produces a head rise of 2117 ft of oxygen (O_2) at a flow rate of 2907 gpm for a rotor speed of 8572 rpm at rated conditions.

6. A motor-driven, propellant utilization (PU) valve (Fig. 7), mounted on the oxidizer turbopump, which bypasses LO_2 from the discharge to the inlet side of the oxidizer pump to ensure simultaneous depletion of propellants.

7. Oxidizer and fuel bleed valves allow trapped gas to be expelled from the engine propellant system before ES. These valves permit the propellant recirculation flow to return to the stage propellant tanks and are closed at ES.

8. A gas generator (GG) (Fig. 8), which consists of a combustion chamber containing two spark plugs, a valve which controls the oxidizer and fuel poppets, and an injector assembly. The high energy gases produced by GG are routed to the fuel turbine, through the turbine crossover duct to the oxidizer turbine, and are exhausted through eyelets into the thrust chamber at an area ratio (A/A_t) of approximately 11.

9. A pneumatically actuated, oxidizer turbine bypass valve (OTBV). At engine start, OTBV is fully open, routing a large portion of fuel turbine discharge gas directly to the thrust chamber to obtain the desired oxidizer-fuel turbine spinup relationship. During engine transition to main stage, OTBV is closed (the valve gate contains a flow nozzle which provides a turbine power balance mechanism).

10. An integral, high pressure gaseous hydrogen (GH_2) start tank and helium (He) control bottle. A pneumatically actuated, normally closed, start tank discharge valve (STDV), controlled by a solenoid-operated valve, permits release of the start tank GH_2 for turbine spinup during the ES cycle. The He control bottle, located within the H_2 start tank, provides a high pressure He supply to the engine pneumatic control system. The start tank is refilled (from fuel injector and manifold supplies) during 60 sec of engine main-stage operation to provide restart capability.

11. A pneumatically actuated, main fuel valve which is a normally closed butterfly-type valve.

12. A two-stage, main oxidizer valve (MOV), which is a pneumatically actuated, normally closed, butterfly-type valve. The first-stage actuator positions MOV at the 14-deg position to obtain initial main chamber ignition; the second-stage actuator ramps MOV full open to accelerate the engine to main-stage operation. The MOV gate is pivoted off-center (Fig. 9), which provides MOV hydraulic torque in the closing direction at the 14-deg position.

13. A pneumatic control package which controls all pneumatically operated engine valves and purges.

14. An electrical control package which provides the electrical logic required for proper sequencing of engine components during operation. It also supplies power to the GG and ASI spark plugs.

15. Primary and auxiliary flight instrumentation packages which environmentally protect and contain sensors required to monitor critical engine parameters.

2.1.2 S-IVB Stage

The S-IVB static battleship stage is approximately 22 ft in diameter and 49 ft long, having a maximum capacity of 46,000 lb of LH_2 and 199,000 lb of LO_2. The major components of the S-IVB stage are (1) propellant tanks, fuel above oxidizer, separated by a common bulkhead, (2) propellant pre-valves which serve as emergency engine shutoff valves and are normally closed during the recirculation chilldown procedure, (3) propellant low pressure ducts which, externally to the tanks, route propellants to the engine pump inlets, (4) propellant recirculation systems which circulate the propellants through the low pressure ducts and turbopumps to stabilize pump temperatures near normal operating levels and prevent temperature stratification in the propellant tanks before ES, (5) vent and relief valve systems for both propellant tanks, and (6) He storage system within the fuel tank (sealed and not utilized for testing at AEDC).

2.2 TEST CELL

Test cell J-4, Fig. 10, is a vertically oriented test unit designed for static testing of large liquid-propellant rocket engines and propulsion systems at pressure altitudes of 100,000 ft. The cell is currently capable of testing engines in the 500,000-lb_f-thrust class (maximum capability is

1,500,000 lb_f). The cell consists of four major components (1) test capsule, 48 ft in diameter and 81 ft in height, situated at grade level and containing the test article, (2) spray chamber, 100 ft in diameter and 250 ft in depth, located directly beneath the test capsule to provide exhaust gas cooling and dehumidification, (3) coolant water, steam, GN_2 and LN_2, GH_2 and LH_2, LO_2, and He storage and delivery systems for operation of the cell and test article, and (4) control building, located about 600 ft from the cell, containing test article and cell controls and data acquisition equipment. Exhaust machinery is connected with the spray chamber. This machinery maintains the test capsule at a pressure altitude of approximately 60,000 ft during the test period, except during engine firing. During firing operations, the facility steam ejector, in conjunction with the exhaust machinery, provides a pressure altitude of 100,000 ft in the test capsule. A detailed description of the test cell is presented in Ref. 3.

The S-IVB battleship stage was installed on a support stand within the test capsule (Fig. 11), orienting the J-2 engine vertically downward on the centerline of the diffuser-steam ejector assembly. This assembly consists of a 20-ft-diam diffuser duct, 150 ft in length, containing a center-body steam ejector. At the inlet to the diffuser is a 13.5-ft-diam diffuser insert, directly above which is a GN_2 annular ejector. The annular ejector was provided to suppress steam recirculation into the test capsule during steam ejector shutdown. The test cell was also equipped with (1) a GN_2 purge system for continuously inerting the normal air in-leakage of the cell; this purge is introduced at the top of the test capsule, (2) a GN_2 repressurization system for rapid emergency inerting of the capsule; this is also introduced at the top of the test capsule, and (3) a spray chamber LN_2 supply and distribution manifold for initially inerting the spray chamber and exhaust ducting and for increasing the molecular weight of the H_2-rich exhaust products during engine operation.

2.3 INSTRUMENTATION

Instrumentation systems were provided to measure engine, stage, and facility parameters; a parameter listing is presented in Table IV. The locations of selected engine and stage instrumentation used during this test series are presented in Fig. 12. The engine instrumentation was comprised of (1) flight instrumentation for the measurement of critical engine parameters and (2) facility instrumentation, which was provided to verify the flight instrumentation and to measure additional engine parameters. The flight instrumentation was provided and calibrated by the engine manufacturer; facility instrumentation was initially calibrated and periodically recalibrated at AEDC.

Pressure measurements were made using strain-gage-type pressure transducers. Temperature measurements were made using resistance temperature transducers (RTT) and a combination of copper-constantan, iron-constantan, and Chromel®-Alumel® thermocouples. Oxidizer and fuel turbopump shaft speeds were sensed by magnetic pickups. Fuel and oxidizer flow rates to the engine were measured by turbine-type flowmeters which are an integral part of the engine. The propellant recirculation flow rates were monitored with turbine-type flowmeters provided in the supply lines by the S-IVB stage manufacturer. Engine side loads were measured with dual-bridge, strain-gage-type load cells which were laboratory calibrated before installation. Vibrations produced during engine operation were measured by accelerometers mounted (in the vertical plane) on the oxidizer dome and (in the horizontal planes) on the turbopumps. Primary engine and stage valves were instrumented with linear potentiometers and limit switches.

The data acquisition systems were calibrated by (1) precision electrical shunt resistance substitution for the pressure transducers, load cells, and RTT units, (2) voltage substitution for the thermocouples, (3) frequency substitution for shaft speeds and flowmeters, and (4) frequency-voltage substitution for accelerometers.

The types of data acquisition and recording systems used during this test period were (1) a multiple-input digital data acquisition system (MicroSADIC®, scanning each parameter at 40 samples per second and recording on magnetic tape, (2) single-input, continuous-recording FM systems recording on magnetic tape, (3) photographically recording galvanometer oscillographs, (4) direct inking, null-balance, potentiometer-type X-Y plotters and strip charts, and (5) optical data recorders. Applicable systems were calibrated before each test (atmospheric and altitude calibrations). Television cameras, in conjunction with video tape recorders, were used to provide visual coverage during an engine firing, as well as replay capability for rapid examination of unexpected events.

SECTION III
CONTROL LOGIC

Control of S-IVB battleship stage, J-2 engine, and test cell systems during the terminal countdown was centrally provided from the test cell control room. The less critical facility, stage, and certain J-2 engine functions were manually controlled. Other functions were programmed to the facility countdown sequencer which provided (1) verification of the readiness of critical systems to proceed with an engine firing and (2) a necessary time display for integrating the manual operations into the countdown.

The critical engine and stage operations were controlled by a facility logic network, which interconnected the required systems to safely start and shut down the engine. The facility control logic was activated at T - 0.5 sec (sequencer time) by the sequencer. The facility and engine controls are briefly described in the following sections.

3.1 FACILITY CONTROL LOGIC

The facility logic was an electrical control network designed to interconnect the engine control system, major stage systems, the engine safety cutoff system (ESCS), observer cutoff circuits, and the countdown sequencer. A diagram of the facility control logic is shown in Fig. 13. The primary functions normally performed by the facility logic were to:

1. Ascertain facility and engine systems ready to test,
2. Open stage propellant prevalves,
3. Shut off stage propellant recirculation pumps and close recirculation valves,
4. Apply start signal to engine control logic, and
5. Initiate facility systems shutdown at expiration of sequencer-programmed run duration; this involved closing the prevalves and initiating facility-supplied engine purges.

The countdown sequencer was programmed to function with the facility logic as follows:

1. At T - 1 sec, verify systems ready, or stop countdown,
2. At T - 0.5 sec, apply firing command to facility logic, and
3. At T - 0 sec, stop sequencer countdown until either (a) the facility logic started the engine to yield STDV solenoid energized or (b) an engine safety cutoff was obtained; if (a), the sequencer resumed counting for the preset length of run and applied an engine cutoff signal at expiration of run duration as well as initiated facility systems shutdown sequence; if (b), the sequencer initiated facility systems shutdown sequence.

The time between fire command and ES varied (as a function of prevalve opening time); it was nominally 8 sec.

A modification to the facility logic was performed to meet requirements to simulate the ES sequence on S-IVB/S-V (flight 501). This modification, called the "auxiliary" logic, was to simulate the following flight sequence:

Time, sec	Event
T_4	S-II/S-V Engine Cutoff
$T_4 + 0.2$	Command S-IVB/S-V Prevalves Open
$T_4 + 1.0$	S-IVB/S-V Engine Start
$T_4 + 1.4$	Shutdown Oxidizer Recirculation Pump
$T_4 + 2.2$	Shutdown Fuel Recirculation Pump

The modifications were accomplished (Fig. 13) and utilized on tests 17A and 17C. A test safety feature was also provided for the auxiliary logic to prevent STDV opening, if both stage prevalves were not fully open (the signal to energize the STDV solenoid produced an automatic engine cutoff in this case).

Automatic engine cutoff circuitry was provided in the facility logic (sequence monitor or start "OK" timer expiration) as well as in ESCS. The ESCS monitored engine vibration and gas generator outlet temperature (GGOT). Engine vibration, sensed by accelerometers mounted on the oxidizer dome, was required to sustain a level equal to or greater than ±150 g for 150 msec to produce an engine cutoff. The GGOT was required to exceed 2000°F (effective 0.8 sec after main-stage solenoid energized) to produce an engine cutoff. This limit was changed to 2200°F (effective 0.7 sec after main-stage solenoid energized) before test 17. An engine cutoff was also produced if GGOT (1) exceeded 1450°F, effective 3.5 sec after main-stage solenoid energized or (2) failed to achieve 250°F by 0.8 sec after main-stage solenoid energized.

3.2 ENGINE SEQUENCE

3.2.1 Engine Start Sequence

An operating sequence diagram and an engine schematic are presented in Figs. 14 and 15. Initiation of ES command (facility-initiated) activates the ES module, which simultaneously opens the He control valve, the ignition phase control valve, and energizes the ASI and GG spark plug exciters. The STDV control and fuel lead timers are also energized at ES command. The He control valve fills the pneumatic accumulator, closes the propellant bleed valves, and purges the oxidizer dome and GG oxidizer injector manifold through the purge control valve, opens the ASI oxidizer valve and the main fuel valve, and supplies pressure to the inlet port of the sequence valve located within the MOV first-stage actuator. With the ASI oxidizer valve and the main fuel valve open, propellants flow under static head to the ASI chamber and are ignited. Once the main fuel valve is 90 percent open, a sequence valve supplies opening pressure to the STDV solenoid control valve.

A normal engine sequence will continue with the opening of STDV and the energizing of the ignition phase timer (450-msec timer), if the following conditions exist (1) the main fuel valve and fuel sequence valve are open, (2) proper fuel quality at the injector is verified by a fuel injection temperature below -150°F, (3) STDV control timer has expired (640-msec timer initiated at ES), and (4) the fuel lead timer has expired. With these four conditions satisfied, STDV opens to release GH_2 to the fuel and oxidizer turbines. Once the ignition phase timer has expired, STDV is closed by de-energizing the control solenoid, a 3.3-sec spark plug de-energize timer is activated, and the main-stage control module is energized. If ASI ignition has not been detected (ASI ignition detect probe), upon expiration of the ignition phase timer, engine cutoff will occur. After the main-stage control module is energized, the main-stage control valve opens, venting He pressure from the MOV closing actuator and the opening port of the purge control valve. The purge control valve closes, and the oxidizer dome and GG oxidizer purges are terminated. Opening pressure is applied to the MOV actuators, and the MOV first stage opens. A sequence valve in MOV supplies pressure to open the GG control valve and to close OTBV. Fuel and oxidizer flow to GG are controlled by poppets in the GG control valve that open sequentially to provide a fuel lead. Gases generated are directed in series to the fuel and oxidizer turbines. The second stage of MOV is sequenced to start opening approximately 0.6 sec after the main-stage control valve is opened. The second-stage valve ramp time is controlled by venting closing pressure through an orificed check valve. As the propellant turbopumps approach steady-state operation, the "main-stage OK" signal is generated by an oxidizer injector pressure switch, and steady-state engine operation follows. If the main-stage OK signal has not been initiated before expiration of the sparks de-energize timer, engine cutoff will occur. The time from ES command to main-stage OK signal is primarily dependent upon the fuel lead time; the relative time between engine starting events may be obtained from Fig. 16a.

3.2.2 Engine Shutdown Sequence

A cutoff signal simultaneously de-energizes the control solenoids for closing of the main-stage and ignition phase control valves and energizes the He control de-energize timer. Opening control pressure for the MOV actuator, ASI oxidizer valve, and main fuel valve is vented. Pressure is supplied to close MOV, to open the purge control valve, to close the ASI oxidizer valve, to close the main fuel valve, and to open the fast shutdown control valve. Oxidizer dome and GG oxidizer line purges begin upon decay of the thrust chamber and GG pressures below the He control pressure. With the exception of the normally open ASI oxidizer valve, propellant bleed valves, and OTBV, all valves are normally closed. Expiration of the He control de-energize timer closes the He control valve, venting control system pressure through the oxidizer dome and GG oxidizer purge lines. Once He control pressure decays to actuation pressure of the purge control valve, the valve closes to stop the purges. Closing pressure to the propellant bleed valve is bled off, and these valves open under spring pressure. The engine cutoff sequence is presented in Fig. 16b.

SECTION IV
PROCEDURE

Pre-operational procedures were begun several hours before each test. All consumable storage systems were replenished, and engine inspections and leak checks were conducted. Propellant tank pressurants and engine pneumatic and purge gas samples were taken to ensure that test specifications were met. (Chemical analysis of propellants was provided by the propellant suppliers.) Facility sequence, engine sequence, and engine abort checks were conducted within a 24-hr time period before an engine firing to verify the proper sequence of events. The abort checks consisted of electrically simulating engine malfunctions to verify the occurrence of an automatic engine cutoff signal. Engine and facility sequence checks consisted of verifying the timing of all engine and facility valves and events to be within specified limits. Engine drying procedures recommended by the manufacturer were performed. A final engine sequence check was conducted immediately preceding each test period.

Oxidizer dome, GG oxidizer injector, and thrust chamber jacket purges were initiated before evacuating the test cell (engine purges required for a typical test period are presented in Table V). Upon completion of instrumentation calibrations at atmospheric conditions, the test cell was evacuated to approximately 0.5 psia with the exhaust machinery, instrumentation calibrations at altitude conditions were conducted, and a cell air in-leakage evaluation was subsequently performed. Immediately before loading propellants on board the vehicle, the cell and exhaust ducting atmosphere was inerted with approximately 20,000 lb of N_2 to reduce the O_2 content to less than 4.9 percent by volume (minimum O_2 required to sustain combustion). At this same time, the cell

N_2 purge was initiated at a rate equivalent to the cell air in-leakage multiplied by 3.2. This cell purge (6- to 10-lb/sec) continuously inerted the air leaking into the cell for the duration of the test period. The vehicle propellant tanks were then loaded to the 30-percent level (a test safety maximum), and the remainder of the terminal countdown was conducted. A typical terminal countdown is presented in Ref. 1.

Engine restart tests were accomplished by conducting a first burn test with a PU valve excursion to 33.3 deg for conditioning the turbine hardware at engine shutdown to predicted flight temperatures and (1) restarting the engine at a specific time after first burn engine cutoff (time determined from previously obtained engine temperature data) or (2) restarting the engine at a time determined (by observation of turbine crossover duct temperatures after first burn engine cutoff) to achieve crossover duct temperature requirements at ES. The restart test requirements necessitated dry, low pressure propellant ducts before initiation of propellant recirculation. The He purges were connected to both fuel and oxidizer low pressure ducts for drying immediately after first burn engine cutoff. Normally, however, the ducts were dry within 15 min after first burn cutoff, and the duct purges were not required. During this test series, turbine crossover duct, closing control line to the MOV second-stage actuator, and pneumatic control package, low temperature conditioning became a requirement on some firings. The crossover duct was chilled internally by the introduction of cold He through an instrumentation fitting; the MOV closing control line and the pneumatic package were externally chilled with He. Conditioning with these systems normally began about 1 hr before ES; the conditioning systems were shut down 30 to 60 sec before ES.

SECTION V
RESULTS AND DISCUSSION

The initial objectives of this test program were to (1) verify J-2 engine start performance at thermal conditions, simulating first burn and restart applications and (2) verify J-2 engine performance at a pressure altitude of 100,000 ft.

Specific test and ES requirements (Table VI) were generated on a test-to-test basis by NASA/MSFC because of the continued unexpected engine performance obtained at AEDC (Ref. 1). The results of the second test in this series produced doubt that the J-2 engine could successfully restart one orbit after first burn cutoff. Therefore, NASA declared Saturn V flight 501 would not be launched until engine restart at simulated orbital conditions had been proved at AEDC. The primary test objectives were to identify the restart problem, to investigate solutions, and to prove successful engine restart based on solution(s) investigated for flight 501 mission requirements. To support attainment of these objectives, an accelerated test schedule was initiated at AEDC on January 9, 1967, and was concluded March 16, 1967. The tests reported herein were obtained between December 2, 1966, and February 5, 1967.

The results presented in this report emphasize the J-2 engine restart problem and investigations into solutions to that problem. First burn comparisons to flight data are also reported. Main-stage performance data are presented and compared to acceptance and nominal engine performance.

5.1 TEST SUMMARY

During this test series, a total of 14 ES were made with firing durations ranging up to 50 sec for an accumulated engine operation time of 293.6 sec. Pressure altitudes at ES ranged from 93,000 to 111,000 ft (geometric altitude, Ref. 4); minimum pressure altitudes during ES transients varied from 79,000 to 96,000 ft. Specific test objectives and a brief summary of results obtained for the firings of S-IVB/S-V tests 12 through 19 are presented in the following table.

Test Objectives		Results
12A-First Burn	Conditions were such as to be conductive to a fuel pump stall.	No pump stall tendencies were noted.
12B-First Burn	Evaluate the effect of low thrust chamber resistance on GG overtemperature.	High GGOT peak (2080°F) with no second peak or GG overtemperature cutoff.

13A-First Burn	Further evaluate contributing factors to GG overtemperature. Determine if 1-sec fuel lead was sufficient with prechilled thrust chamber. Condition turbine hardware temperature for restart.	Low GGOT (1782°F); 1-sec fuel lead was sufficient at this thrust chamber temperature.
13B-Restart	Evaluate the start characteristic at simulated one orbit conditions. Comparison to 10B, but with much warmer turbine hardware temperatures.	Engine cutoff from a very high GGOT (2426°F). The MOV did not move off the 14-deg plateau.
14-First Burn	To clear first burn (flight) for a 1-sec fuel lead. Evaluate the effects of conditioning towards a fuel pump stall.	Fuel lead of 1 sec was not sufficient to condition engine for start. No pump stall tendencies were noted.
15A-First Burn	A repeat of 10B and to condition engine for restart.	Results were similar to 10B with a maximum GGOT of 1799°F.
15B-Restart	Evaluate the effects of restarting at a PU valve setting of 22 deg on the start transient. This was a worst-case GG overtemperature, repeating 13B, with the exception of PU valve position.	Although GG overtemperature cutoff occurred, a significant reduction in GGOT from 13B (2426 to 2132°F) resulted. Also, MOV had moved off the 14-deg plateau before engine cutoff.
15C-First Burn	Repeat of 12B with worst-case GG overtemperature for first burn.	A high GGOT peak (2071°F), but no cutoff occurred.
16A-First Burn	Further evaluate the effects of the PU valve position on the start transient.	A significant reduction in GGOT (1753°F), but this temperature was influenced by the inadvertent chilling of the turbine crossover duct.

17A-First Burn	Evaluate effect of crossover duct at expected boattail environment (cold duct) on engine buildup time. Evaluate effect of flight 501 prevalve sequencing and reduced fuel lead (2.5-sec). Establish first start similarity between AEDC and flight operation. Comparison to test 15C to establish chilled crossover duct gain on oxidizer turbopump spin speed.	Buildup time appeared normal for tests conducted at AEDC. Prevalve sequencing improper. Initial oxidizer turbopump spin speed was 3437 rpm as compared to 3461 on test 15C.
17B-Restart	Evaluate effect of PU valve open during start at Saturn V (flight 501) maximum crossover duct temperature and start energy (worst-case GGOT). Establish feasibility of PU valve open for flight 501. Comparison to test 17A to establish the chamber conditioning effects.	No cutoff occurred, but a high GGOT (2176°F) was recorded (see Section 5.2.2).
17C-First Burn	Evaluate blowdown stall margin with warmest expected thrust chamber, reduced fuel lead, and flight 501 prevalve sequencing. Comparison to test 17A to establish chamber conditioning effects and post-fire coast temperature data following a shutdown at a PU valve setting of -22 deg.	Pump did not approach stall. Warmer thrust chamber on 17C resulted in a reduction of approximately 100°F in maximum GG temperature. The ASI ignition detect probe failed to de-energize at conclusion of firing. Turbine crossover duct temperatures were approximately 80°F cooler than 17A, 10 min after engine cutoff.

18A-First Burn	Establish maximum buildup time with PU valve in the open position. Comparison to test 16A to establish start tank gain factors with PU valve in open position.	Slowest chamber pressure buildup experienced in this test series (2.365 sec to PC = 550). The ASI ignition detect probe failed to de-energize at conclusion of firing.
19A-First Burn	Establish first burn similarity between AEDC and flight operation.	Comparison to flight data is presented in Section 5.3.4. The ASI ignition detect probe failed to de-energize at conclusion of the firing.

Propellant pump inlet and start tank conditions obtained are compared to safe start envelopes in Fig. 17. Specific test results are summarized in Table VII. Engine valve sequence on all tests (start and shutdown) is presented in Table VIII. Engine MOV pre-test sequence checks are shown in Table IX.

General unexpected observations concerning this test series are as follows:

1. Transient GGOT (initial peak) averaged 1880°F on first burn tests and 1890°F on restart tests. Transient GGOT (second peak) ranged from no second peak to 1840°F on restart tests. Transient GGOT (second peak) ranged from no second peak to 1840°F on first burn tests, averaging 2240°F on the three restart tests. Two of the three restart tests (13B and 15B) were prematurely terminated by ESCS because of excessive GGOT. Both firings were conducted at conditions simulating turbine hardware temperatures after one orbit, based on flight AS 203 data.

2. Second-stage MOV delay time averaged 679 msec (93 msec greater than pre-test sequence checks) on first burn tests and 809 msec (223 msec greater than pre-test sequence checks) on the restart tests (15B and 17B). The MOV second-stage actuator was unable to move the valve on restart test 13B. Pre-test sequence checks of MOV on all tests were within specifications; second-stage actuation time averaged 586 msec (delay) and 1803 msec (ramp).

3. Excessive engine vibration occurring at approximately T + 1 sec (oxidizer dome prime) was experienced on 13 of the 14 ES, measured vibration safety count (VSC) duration ranged up to 90 msec and averaged 28 msec.

4. The ASI restartable ignition detect probe failed to de-energize after firings 17C, 18A, and 19A, resulting in termination of the scheduled subsequent firings. Inspection after each of these tests has shown the probe to be damaged by excessive ASI combustion temperatures. The severity of the ASI combustion temperatures on test 19 resulted in injector face damage as shown in Fig. 18. The start conditions for these tests combined a low fuel pump inlet pressure with a high oxidizer pump inlet pressure. It is surmised that these probe failures were a result of a high ASI oxidizer to fuel ratio during the fuel lead.

5. Test 16 was terminated after one firing because of a suspected ASI propellant leak and a malfunction of the turbine crossover duct conditioning system. The ASI assembly was replaced between tests 16 and 17.

6. First burn (flight 501) sequence was utilized on tests 17A and C. However, prevalve opening time was faster than the planned flight sequence, and therefore, this test objective was not obtained.

5.2 ENGINE RESTART AT SIMULATED ORBITAL CONDITIONS

5.2.1 Restart Problem

The premature termination of test 11B (Ref. 1) by ESCS because of excessive GGOT (2150°F) led to a restart investigation on subsequent tests. During the ES transient, any condition that increases the normal GG oxidizer injector pressure buildup rate or decreases the normal GG fuel injector pressure buildup rate will increase the gas generator O/F ratio, causing the temperature to increase above the normal operating temperature. Test conditions for test 11B included an 8-sec fuel lead and warm turbine hardware temperatures, both of which were suspected to contribute to the excessive GG temperature because of their effect on these injector pressures. The 8-sec fuel lead produces a very cold thrust chamber at altitude (Fig. 19), which reduces the resistance to fuel flow through the thrust chamber and results in a lower GG fuel injection pressure. Figure 20 presents an indication of this resistance for test 11B. The turbine hardware temperatures for test 11B were obtained by restarting the engine 1 hr, 35 min after test 11A. The turbine hardware temperatures at T_O for test 11B are presented in the following table.

Parameter	Temperature, °F
Oxidizer Turbine Inlet (TOTI)	136
Oxidizer Turbine Outlet (TOTO)	196
Fuel Turbine Inlet (TFTI)	68
Turbine Crossover Duct (TFTD-1)	138
Turbine Crossover Duct (TFTD-3)	93
Turbine Crossover Duct (TFTD-4)	89

Warm turbine hardware temperatures add energy to the start tank gas as it travels through the turbines and crossover duct during start tank blowdown. This produces an abnormal balance in energy supplied to the turbines, causing a higher than normal oxidizer pump spin rate during the start transient. This increases the GG oxidizer injector pressure and increases the hydraulic torque across MOV. The valve design is such that its second-stage actuator must overcome this torque to move the valve off the 14-deg position and begin the second-stage ramp. A sufficient increase in this torque will delay the beginning of the second-stage ramp. Such a delay is undesirable since the oxidizer pump discharge pressure buildup rate is higher with the valve in the 14-deg position.

Conditions for test 12B were selected to evaluate the effect of low fuel system resistance on the ES transient. Thrust chamber resistance to fuel flow was essentially the same on test 12B as test 11B (Fig. 20), and turbine hardware temperatures were ambient. These conditions produced GGOT of 2080°F with no second peak (Fig. 21). Figure 21 presents a comparison of GGOT obtained on tests 11B and 12B. This figure shows that (1) both tests have very high initial peaks that occur too early to produce an engine cutoff, (2) test 11B had an excessive second peak, and (3) 12B had no second peak. A comparison of these two tests indicates that very low fuel system resistance is the prime contributor to the high initial peaks and warm turbine hardware temperatures produce an excessive second peak.

Test 13B conditions, a repeat of test 10B except for turbine hardware temperatures, were selected to further evaluate the effect of turbine hardware temperatures on the ES transient. Thrust chamber temperatures at T_0 (Fig. 19) for test 13B, given below, were warm enough to give high fuel system resistance (Fig. 20), and turbine hardware temperatures were much warmer than on test 11B (comparable to temperatures experienced on flight AS-203 after approximately one orbit, Fig. 22).

Parameter	Temperature, °F
Oxidizer Turbine Inlet (TOTI)	292
Oxidizer Turbine Outlet (TOTO)	287
Fuel Turbine Inlet (TFTI)	269
Fuel Turbine Outlet (TFTO)	316
Turbine Crossover Duct (TFTD-1)	282
Turbine Crossover Duct (TFTD-3)	202
Turbine Crossover Duct (TFTD-4)	211

Test 13B was terminated at 1.33 sec because of a very high GGOT (second peak of 2430°F, Fig. 23a) verifying that turbine hardware temperatures have a significant effect on the GG start transient. Oxidizer pump speed buildup during the start transient (Fig. 23b) was higher than any experienced on previous tests (Table X). As a result, the oxidizer pump discharge pressure buildup rate was also very high (Fig. 23c). Figure 23d, showing oxidizer and fuel injector pressures, gives an indication of the high O/F ratio being supplied to GG. Also, the oxidizer pump discharge pressure developed such high torque across MOV that the second-stage actuator was unable to move the valve off the 14-deg position. Figure 24 shows the oxidizer pump discharge to chamber pressure differential; this is the best available indication of differential pressure across MOV. The manufacturer suggests the hydraulic torque (in.-lb_f) at the 14-deg position may be obtained by multiplying this differential pressure (psid) by 0.797.

Fuel turbine inspections were made after tests 11B and 13B to determine the effect of the high GG temperatures on the fuel turbine hardware. The inspection revealed very slight erosion of the turbine blades on the leading edge of the first-stage rotor, after test 13 (Fig. 25). However, the manufacturer determined that the turbine was satisfactory for continued use.

5.2.2 Restart Investigation

The results of test 13B indicated the oxidizer pump high spin rate during the start transient was the cause for higher than normal (1) hydraulic torque on MOV and (2) oxidizer pump discharge pressures which led to higher than predicted GGOT. The cause for this high spin rate was the additional energy transferred to the start tank gas by the warm turbine hardware. Several means for decreasing the oxidizer turbine spin rate were considered (1) cooling of the turbine hardware, (2) change OTBV and/or MOV sequence, (3) decrease start tank energy, and (4) open PU valve.

The latter of these was selected for investigation of the restart problem with some emphasis placed on reduced start tank energy level. The open PU valve absorbs more of the horsepower developed by the oxidizer turbine by recirculating more oxidizer back through the oxidizer pump and acting as a hydraulic brake.

The test conditions for test 15B were essentially the same as for test 13B, except the PU valve position was set at -22 deg (Fig. 27). Turbine hardware temperatures for this test are compared to flight AS-203 temperatures in Fig. 26. Temperatures at T_0 for test 15B are tabulated in the following table.

Parameter	Temperature, °F
Oxidizer Turbine Inlet (TOTI)	301
Oxidizer Turbine Outlet (TOTO)	306
Fuel Turbine Inlet (TFTI)	293
Fuel Turbine Outlet (TFTO)	334
Turbine Crossover Duct (TFTD-1)	299
Turbine Crossover Duct (TFTD-3)	214
Turbine Crossover Duct (TFTD-4)	218

Figure 28 shows a comparison of the percentage of the horsepower absorbed by the PU valve in the null position on test 13B to the percentage absorbed at the -22-deg position on test 15B. Test 15B did result in an engine cutoff because of a high GGOT, but a comparison to test 13B shows that the GG temperature peaked at a much lower temperature (2130°F, Fig. 29a), and the oxidizer pump did not spin up as high (Fig. 29b). Also, the rise rates of the propellant pump discharge pressures (Fig. 30a) and the GG injector pressures (Fig. 30b) were not as high, and MOV had moved off the 14-deg position. These test results indicated a significant advantage could be gained from starting with PU valve in the full open position (-29 deg).

After test 15B, the GG outlet maximum temperature cutoff limit was raised from 2000°F, effective 0.8 sec after main-stage solenoid energized to 2200°F, effective 0.7 sec after main-stage solenoid energized. Test data indicate that test 15B would not have exceeded these limits.

Test 17B, which was essentially a repeat of test 11B except the PU valve was full open (-29 deg), did not receive an engine cutoff. Turbine hardware temperatures for this test are compared to flight AS 203 temperatures in Fig. 31. Turbine hardware temperatures at T_0 for test 17B were planned as maximum expected for Saturn V (flight 501) and are tabulated in the following table.

Parameter	Temperature, °F
Oxidizer Turbine Inlet (TOTI)	218
Oxidizer Turbine Outlet (TOTO)	248
Fuel Turbine Inlet (TFTI)	113
Fuel Turbine Outlet (TFTO)	223
Turbine Crossover Duct (TFTD-1)	205
Turbine Crossover Duct (TFTD-3)	166
Turbine Crossover Duct (TFTD-4)	156

The PU valve position used on this test gives the maximum flow rate through the PU valve and yields the highest horsepower absorption by the PU valve (Fig. 28). The effect of this PU valve position on propellant pump discharge and GG injector pressures is shown in Fig. 32. Also, the reduction in oxidizer pump spin speed for this test can be seen by comparison to tests 13B and 15B in Fig. 29. A comparison of GGOT is also shown in this figure.

From this restart investigation, it was apparent that warm turbine hardware temperatures contribute significantly to high GG start transient temperatures. Test 13B demonstrated that the engine would not restart, under normal operating conditions (null PU valve) with turbine hardware temperatures comparable to flight after one orbit, without experiencing excessive GG temperatures sufficient to cause performance degradation and possible engine failure. At engine shutdown on test 13B (T_0 + 1.33 sec), MOV had not begun its second-stage ramp. Test 15B indicated the engine could satisfactorily restart at these hardware temperatures, if the PU valve were moved to the open position. At engine shutdown on test 15B (T_0 + 1.36 sec), MOV had begun its second-stage ramp. Test 17B was a satisfactory restart with an open PU valve and turbine hardware at the highest temperatures expected for Saturn V (flight 501).

5.3 ENGINE START INVESTIGATION FOR SIVB/SV FIRST BURN

The primary objectives of the first burn tests in this series were to (1) investigate the worst-case conditions for fuel pump stall, (2) investigate GGOT during the start transient, (3) investigate fuel lead effects at pressure altitude, and (4) compare AEDC and flight data.

5.3.1 Fuel Pump Stall Investigation

For conditions conducive to pump stall on tests at AEDC, no pump stall tendencies were observed. The minimum stall margin on this series was approximately 850 gpm on tests 14A and 18A. The fuel pump performance is presented in Fig. 33.

It should be noted that the fuel flow for the headflow plots is total pump flow obtained by adding estimated GG flow rate to the measured thrust chamber flow rate. The estimated GG fuel flow rate is based on calculated data from steady-state performance and is 4.5 percent of thrust chamber fuel flow rate.

5.3.2 Gas Generator Temperatures

Although no engine cutoffs caused by excessive GG temperature occurred on first burn tests, the GG transient temperatures were of interest. Test data taken at AEDC had shown the GG transient temperatures at altitude were higher by approximately 500°F than recorded during acceptance tests.

Tests 12A, 13A, 14, 15A, and 15C (first burns) were conducted with an ambient temperature environment; however, data from flight AS 203 (Ref. 5) indicated a thermal environment around the engine of -60 to -80°F at ES. Since subjecting the engine to a thermal environment of this degree would have required major facility modification, it was decided to subject only critical engine components to this environment. As shown in Table V, the turbine crossover duct and associated hardware were preconditioned on first burn simulations, beginning with test 16A. The MOV closing control line and pneumatic control package, as well as the turbine crossover duct, were preconditioned on test 19.

Effects on GG temperature could not be established from this series of of tests as a result of variations in start tank energy, PU valve position, fuel lead time, and thrust chamber conditioning. As discussed earlier (Section 5.2.1) a reduction in turbine crossover duct temperature results in a lower energy addition to the start tank gases during start tank blowdown, which tends to reduce the maximum GG temperature.

5.3.3 Fuel Lead Effects at Altitude

Thrust chamber preconditioning and fuel lead effects at altitude were investigated during this series of tests. As previously discussed, a low fuel system resistance (cold thrust chamber) results in a lower GG fuel injector pressure and, consequently, contributes to the higher GG temperatures. The fuel lead more effectively chills the thrust chamber at altitude than at sea-level pressures. The major differences observed at altitude conditions were (1) a lower back pressure which produces sonic flow through the injector and throat shortly after ES and (2) less heat transfer to the thrust chamber than at sea level as a result of eliminating the sea-level test devices (nozzle diffuser and exit igniters) and convective heating because of air circulation. The effectiveness of the fuel lead at altitude is presented in Fig. 34. In all tests conducted, the fuel injector temperature was below -150°F in less than 5 sec after ES.

5.3.4 Comparison of Flight and AEDC Data

5.3.4.1 Engine Start Transient

One of the primary purposes of tests 17A and 19A was to obtain data at AEDC comparable to flight data. Start requirements for these tests were similar to conditions on Saturn IB flights at ES. The ES conditions for flight AS 201 (Ref. 6), 202 (Ref. 7), 203 (Ref. 5), and AEDC tests 17A and 19 are shown in Table XI. The ES transient data from AEDC tests compare satisfactorily with the limited data available from flight. Comparison of thrust chamber pressure, MOV position, fuel pump discharge pressure, fuel pump performance, and oxidizer pump discharge pressure for tests 17A, 19, and flight are shown in Figs. 35 through 39, respectively.

5.3.4.2 Thrust Chamber Temperatures during Boost-Phase Warmup

A 550-sec boost-phase warmup was conducted on tests 14A and 19. The tests were to simulate thrust chamber warmup for the boost phase of the S-IVB stage first burn on flight 501. These data were compared with flight data obtained on AS 203 in Fig. 40. Although AS 203 had only a 145-sec boost phase, the thrust chamber warmup rates for the throat (Fig. 40a) compare very well with data obtained at AEDC. Warmup rates for AS 201 and 203 were 0.15 and 0.21°F/sec, respectively, compared with rates of 0.1 and 0.12°F/sec obtained at AEDC. The thrust chamber exit temperature comparison (Fig. 40b), however, indicates the exit warmup rate obtained at AEDC is influenced by heat transfer between cell inerting gases (see Section IV) and the thin wall of the thrust chamber. Engine ambient pressure during the 550-sec simulated boost phase ranged from 1.0 psia at T - 550 sec to 0.1 psia at T - 50 sec. Differences in thrust chamber exit warmup rates obtained at AEDC are primarily a result of the removal of the Larodyne insulation from the thrust chamber and installation of the thrust chamber heater blankets between tests 17 and 19.

5.3.4.3 Fuel Lead Effects

A comparison of fuel lead effects for 8-sec fuel leads conducted at AEDC and a 12-sec fuel lead conducted at the end of the first orbit on flight 203 further emphasizes the correlation between altitude testing and flight. The thrust chamber throat (CO 199) and the fuel injector (CO 200) temperatures compare very well between AEDC and flight (Fig. 41). However, the thrust chamber exit temperatures at AEDC do not compare closely with the flight data.

5.4 ENGINE PERFORMANCE

Engine performance data were calculated from test measurements utilizing the PAST 640 computer program, a standard J-2 engine performance program developed and programmed by the engine manufacturer. This program calculates the engine and engine component performance based upon (1) measured data and (2) measured data with pump inlet conditions normalized to standard pump inlet conditions. The required program constants, which included engine dimension measurements, engine flowmeter calibration constants, pump headflow, pump efficiency, and thrust coefficient curve fit constants, were provided by the engine manufacturer. Engine test measurements required by the performance program were obtained from the digital data acquisition system by averaging the 40 data samples obtained in the 1-sec time intervals of interest. Fuel and oxidizer engine flowmeter cyclic output data were manually reduced from oscillograph traces. Pertinent performance program equations and measured data required are presented in Appendix III.

Selected engine performance data computed from measured data (Table XII) and measured data with pump inlet conditions normalized to standard conditions (Table XIII) are presented. Also, for comparison purposes, test 315001 of the acceptance tests on engine S/N J-2052 (at the altitude facility of the engine manufacturer) is included. Performance data from test 315001 are in good agreement with performance data from tests conducted at AEDC.

Programmed engine firing durations during this series of tests at AEDC ranged from 5 to 50 sec. Performance data were computed and are presented for all tests of 30-sec duration or longer. Data from test 12B (50-sec duration) and 15A (40-sec duration) indicate that engine performance has essentially reached steady state after 30 sec of operation at altitude conditions. Plots of thrust chamber mixture ratio, GG mixture ratio, thrust chamber pressure, and characteristic velocity are presented for firings 12B and 15A in Figs. 42 and 43.

Factors limiting the confidence level in the engine performance data are:

1. Engine thrust was not measured, but was calculated from a chamber pressure relationship established during acceptance tests.
2. Redundant propellant flowmeters were not used.
3. Fuel tank repressurant flow was calculated.

The basic engine performance at AEDC is depicted in Figs. 44 and 45. It is noted from these figures that the thrust chamber fuel and GG propellant flow rates are consistently lower than average engine data but compare closely with the acceptance test data. The curves presented represent average J-2 engine performance (Ref. 2).

Performance data indicate there was no significant degradation of turbine efficiencies through this series of tests (Fig. 46). Although the oxidizer turbine performance was not expected to shift, some concern had been expressed over the fuel turbine efficiency because of the high GG temperatures experienced. The fuel turbine inspections after tests 11 and 13 indicated no significant turbine erosion; the performance data support this conclusion.

A shift in fuel pump efficiency was noted after test 13. Data from test 13A compared very well to acceptance test data. At the present time, no explanation can be given for the shift. Figure 47 shows the fuel and oxidizer pump efficiencies for these tests.

SECTION VI
SUMMARY OF RESULTS

Tests 12 through 19 of the J-2 rocket engine (in the Saturn S-IVB stage configuration) were conducted from December 2, 1966, to February 5, 1967, in Propulsion Engine Test Cell (J-4). Pressure altitudes at ES ranged from 93,000 to 111,000 ft. The results of the 14 engine firings of this test series are summarized as follows:

1. Test 12B, a restart with a very low thrust chamber resistance to fuel flow, indicated that the 8-sec fuel lead was not the prime contributor to the GG overtemperature cutoff experienced on test 11B.
2. Test 13B ES occurred with turbine hardware temperatures comparable to those experienced on flight AS 203 after one orbit. As a result, the GG temperature reached 2426°F just before ESCS shut down the engine.
3. On the average, the second-stage MOV delay time exceeded the pre-fire sequence check times by 93 msec on first burn tests and 223 msec on restart tests 15B and 17B. On test 13B, also a restart, MOV failed to begin the second-stage ramp.
4. Test 15B, the first time the J-2 engine had been started at a PU valve setting of -22 deg, resulted in a GG overtemperature cutoff; however, the maximum GG temperature (2132°F) was 294°F lower than on test 13B.
5. Test 17B, a successful restart at a PU valve setting of -29 deg, experienced a maximum GG temperature of 2176°F. Although this temperature is high, the test indicated the engine could be restarted at worst-case turbine hardware temperatures expected on Saturn V (flight 501). Before test 17B, the GG temperature cutoff limit was raised from 2000 to 2200°F.
6. Flight and AEDC data on tests 17A and 19 for engine performance, fuel lead effects, and boost phase warmup compare well.
7. Although engine performance of engine S/N J-2052 at AEDC is low compared to average engine performance, it compares closely with acceptance test data.
8. Performance data from tests at AEDC indicate the engine is essentially at steady state after 30 sec of operation at altitude.
9. Test 14, worst-case pump stall conditions tested during this series, experienced a minimum stall margin of 850 gpm.

10. Vibration safety counts were experienced on 13 of the 14 tests conducted in this series.

11. The combination of low fuel pump inlet pressure and high oxidizer pump inlet pressure resulted in ASI ignition detect probe failure on three tests.

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6. "J-2 Engine Performance on Saturn AS-201." Rocketdyne Division of North American Aviation, Inc. R-6750-1.

7. "J-2 Engine Performance on S-IVB Stage of Saturn Flight AS-202." Rocketdyne Division of North American Aviation, Inc. R-6750-3, January 1967.

APPENDIXES

I. ILLUSTRATIONS
II. TABLES
III. TEST MEASUREMENTS REQUIRED
BY PERFORMANCE PROGRESS

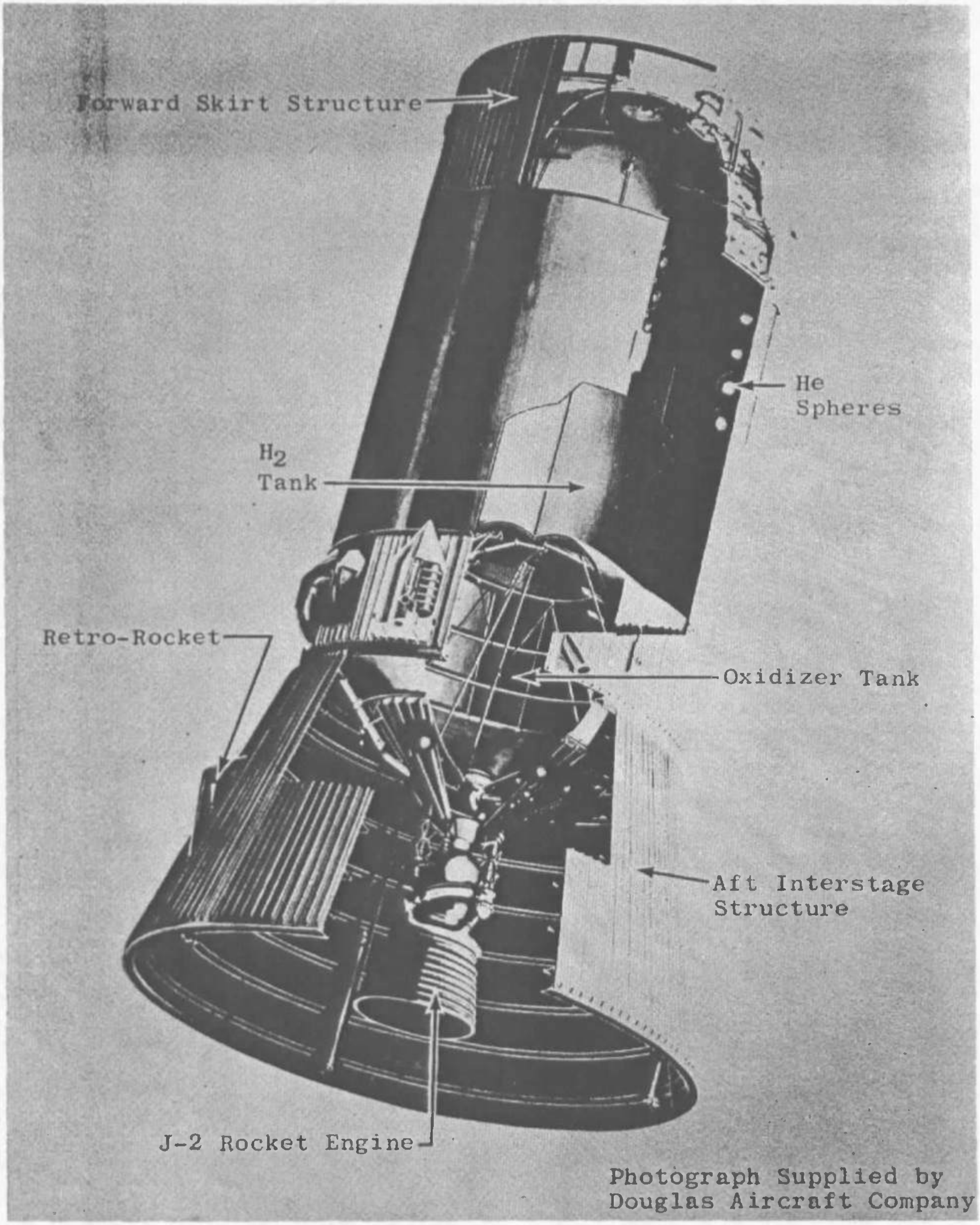
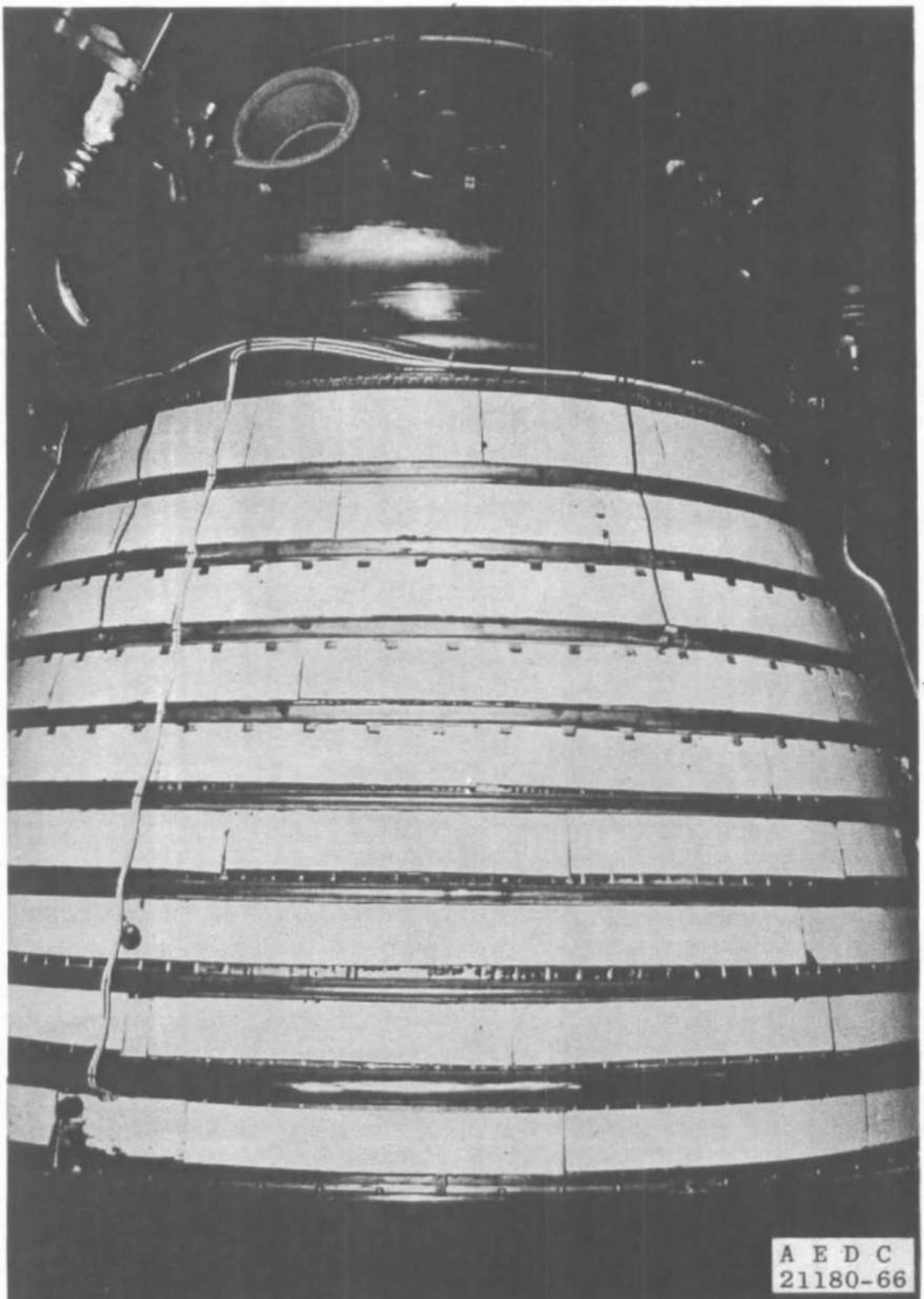
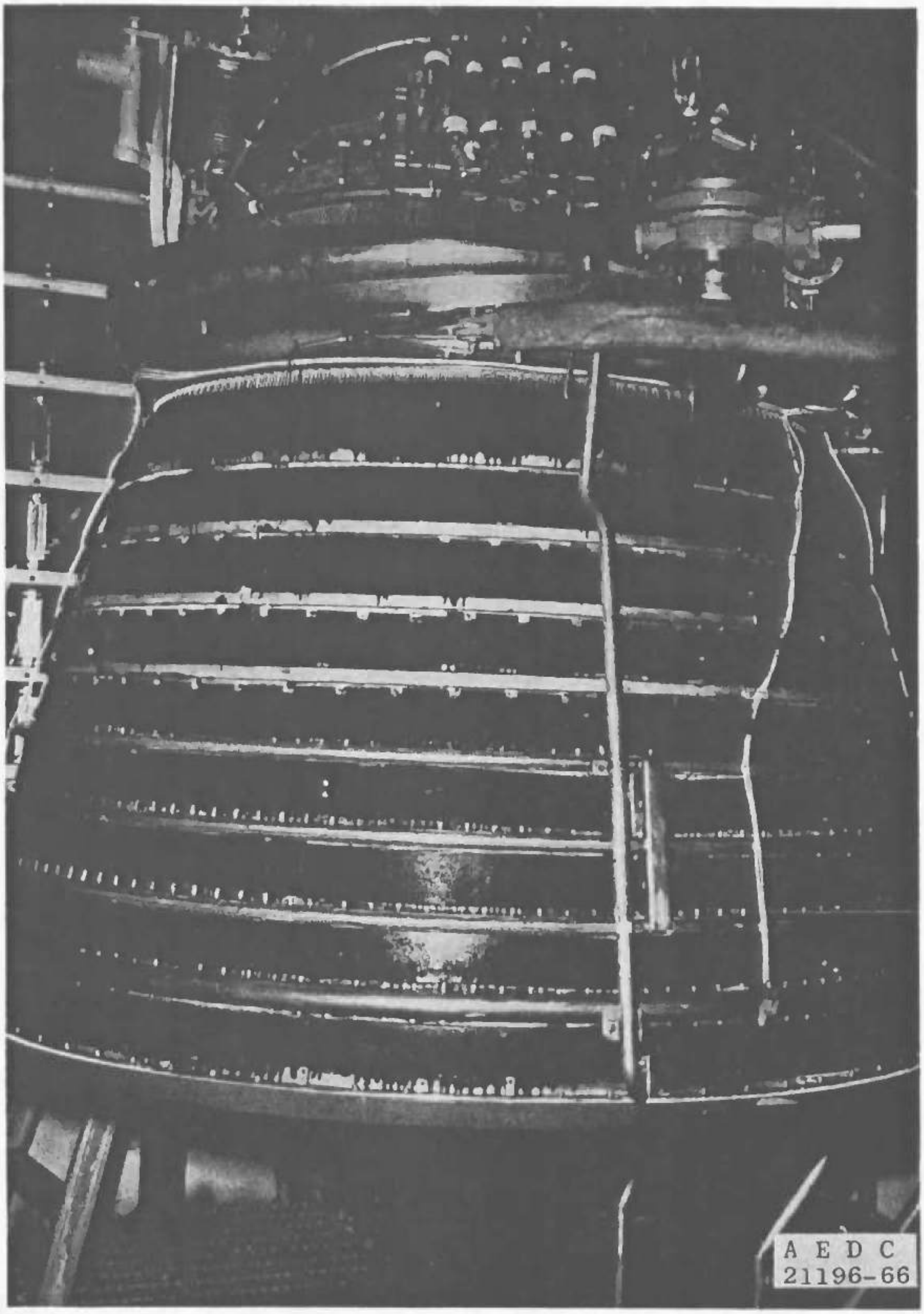


Fig. 1 J-2 Engine and S-IVB Flight Assembly



a. Before Painting

Fig. 2 Thrust Chamber Insulation (S-IVB Configuration)



b. After Painting
Fig. 2 Concluded

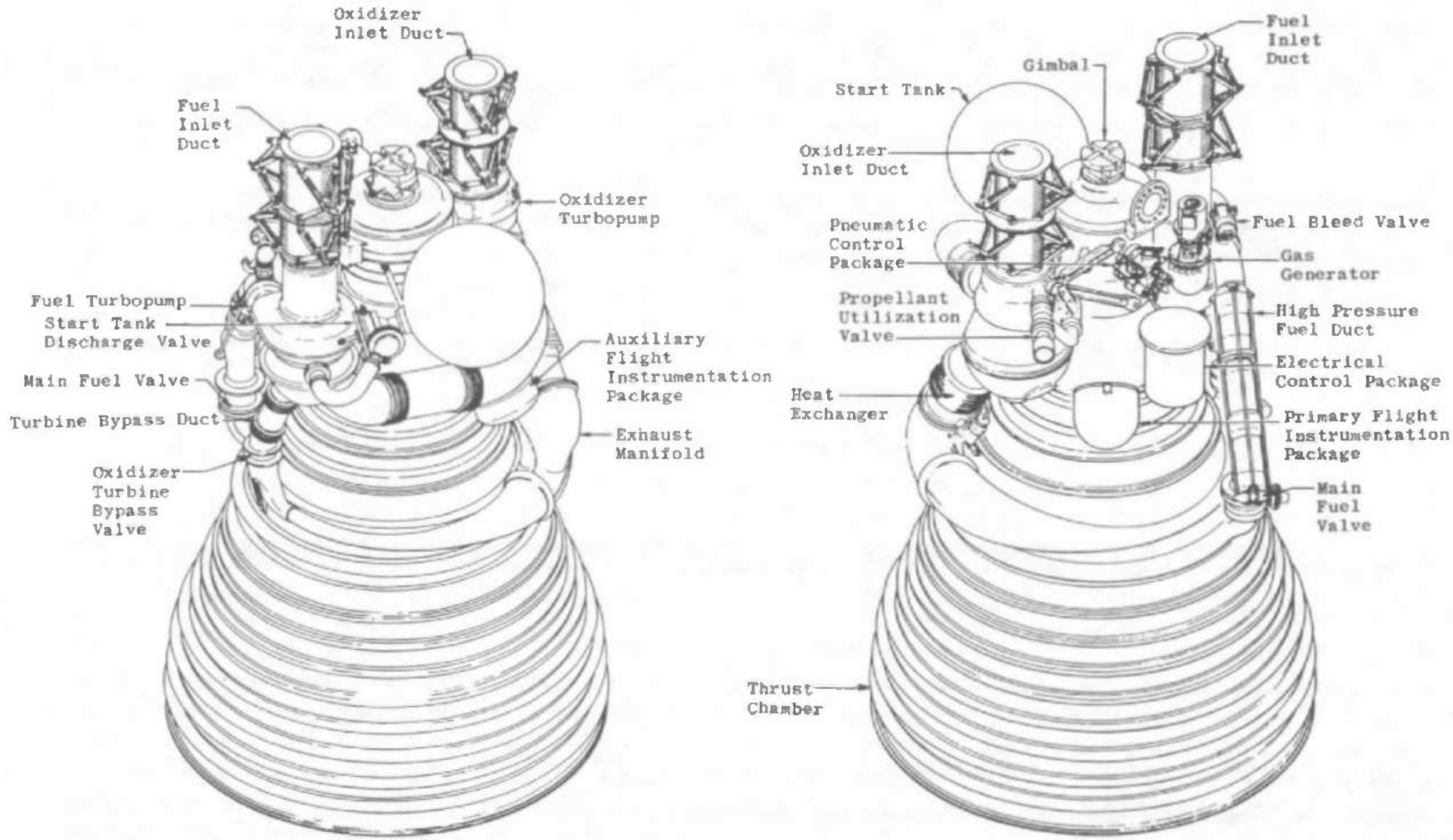


Fig. 3 Details of the J-2 Engine

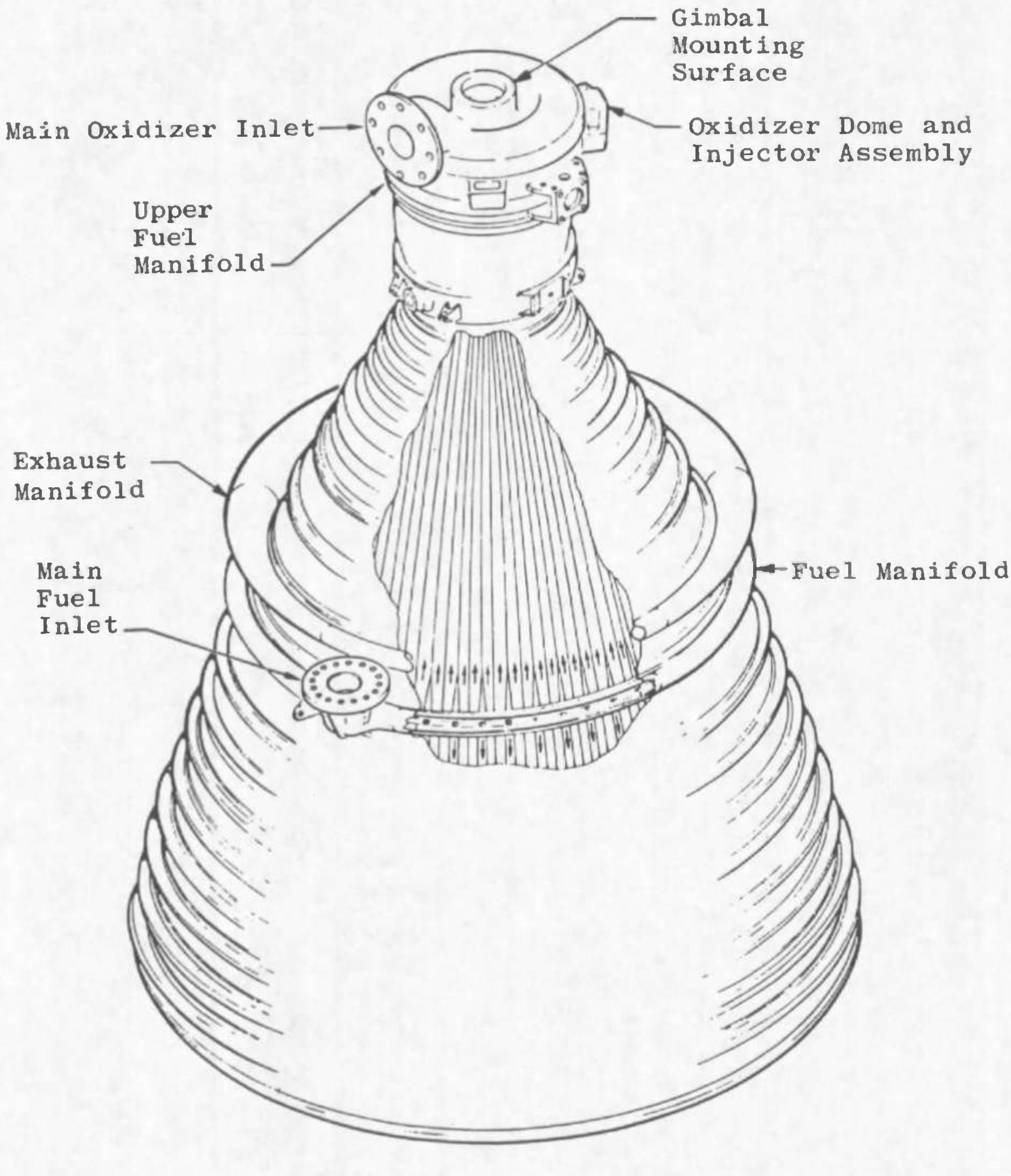


Fig. 4 Details of the J-2 Engine Thrust Chamber

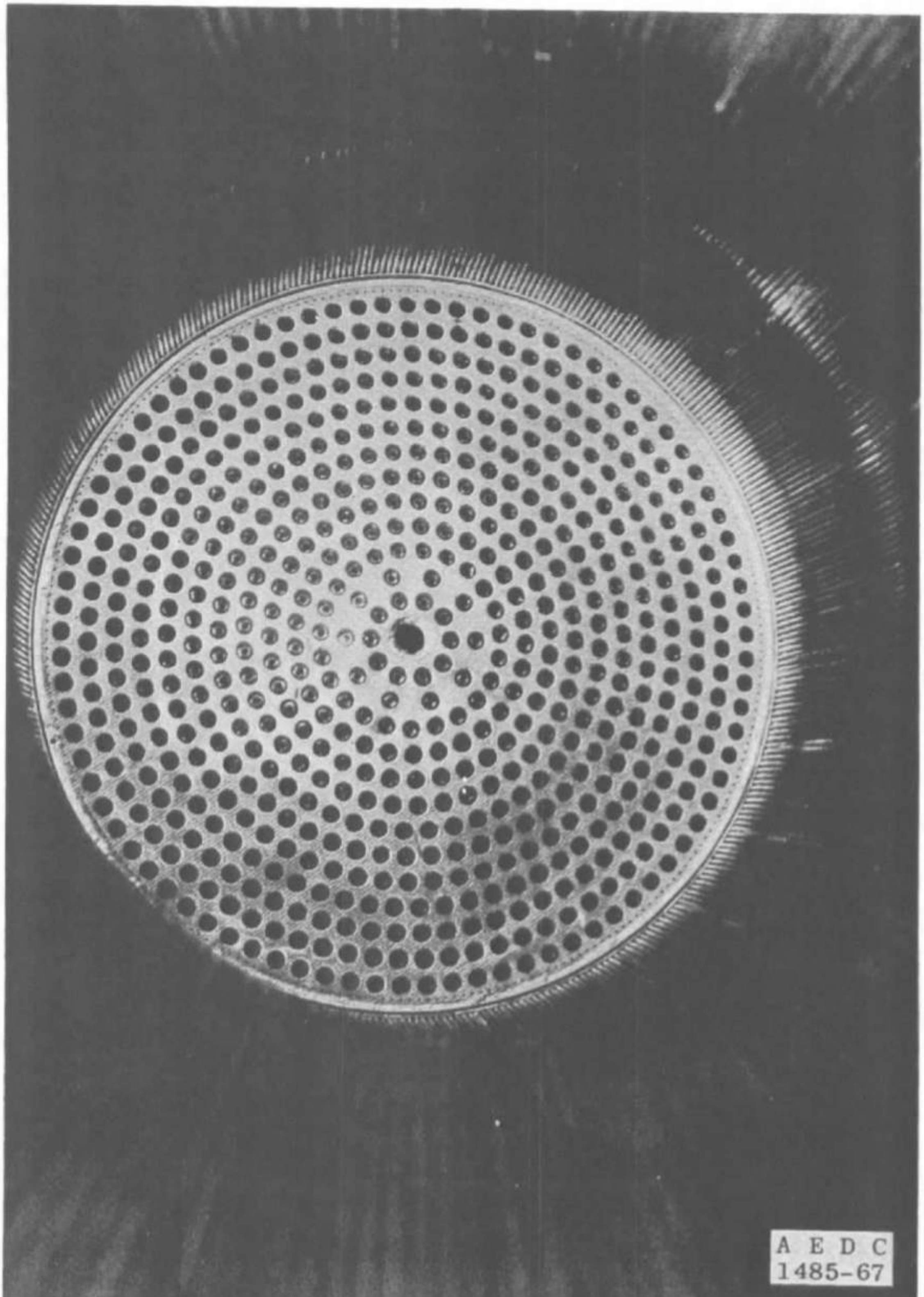


Fig. 5 Details of the J-2 Engine Injector

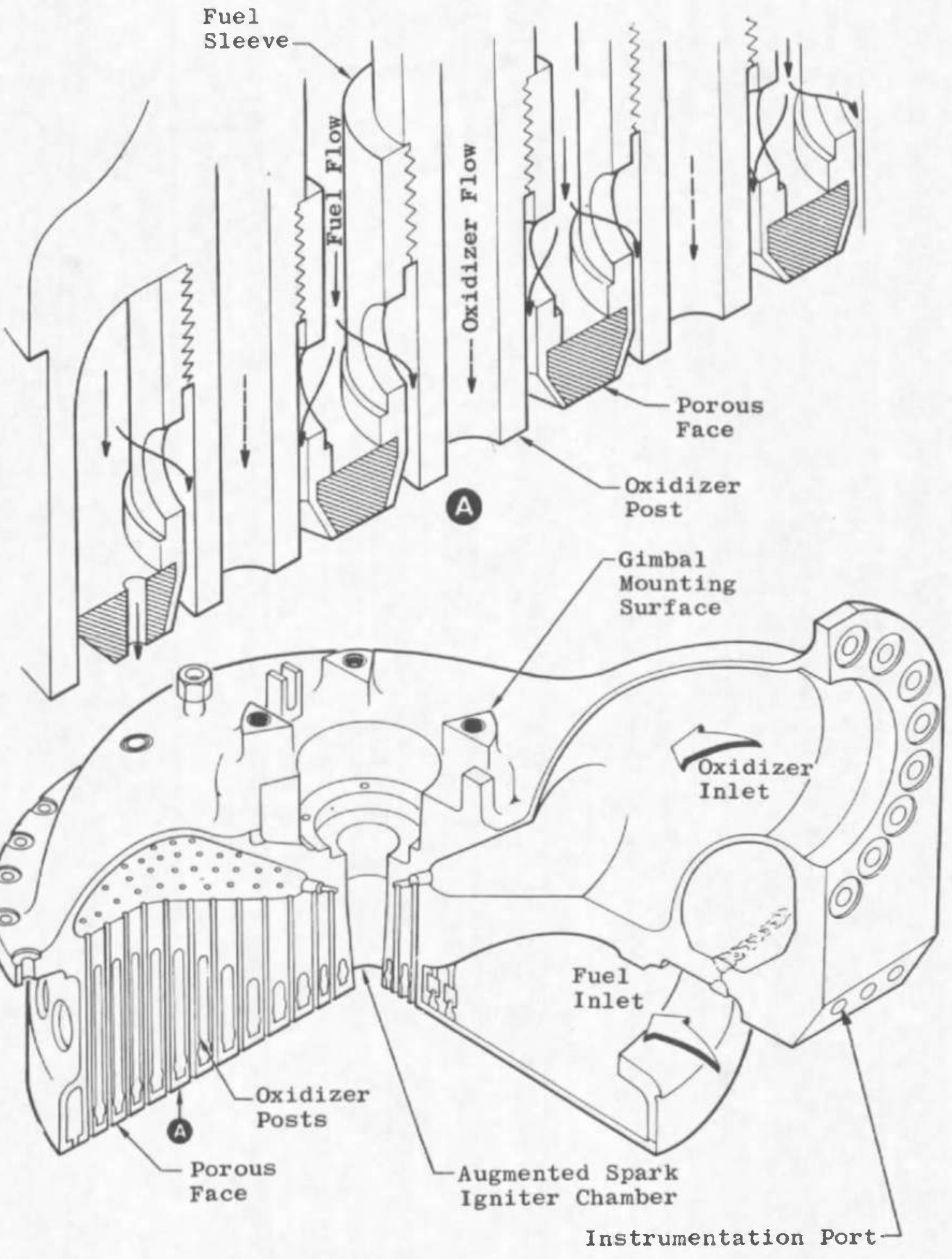


Fig. 5 Concluded

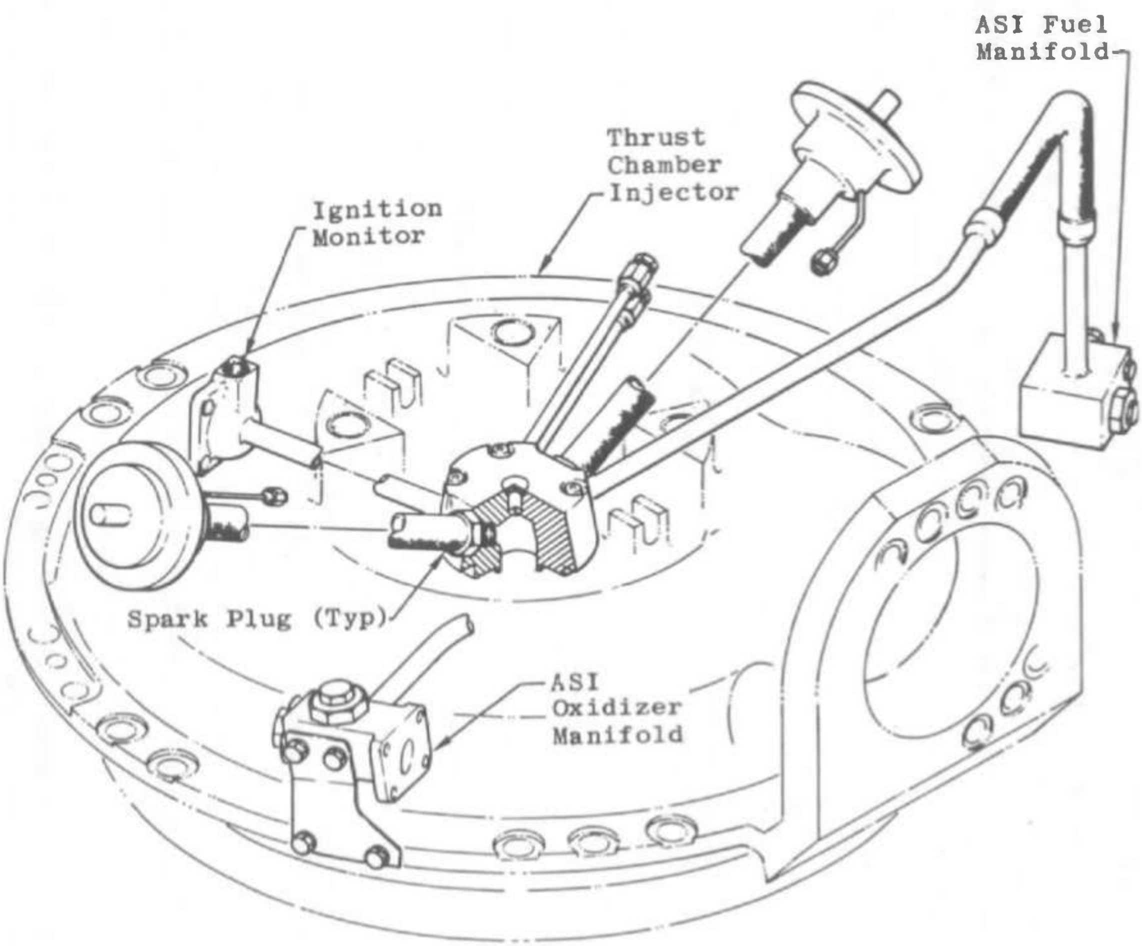


Fig. 6 Details of the ASI Unit

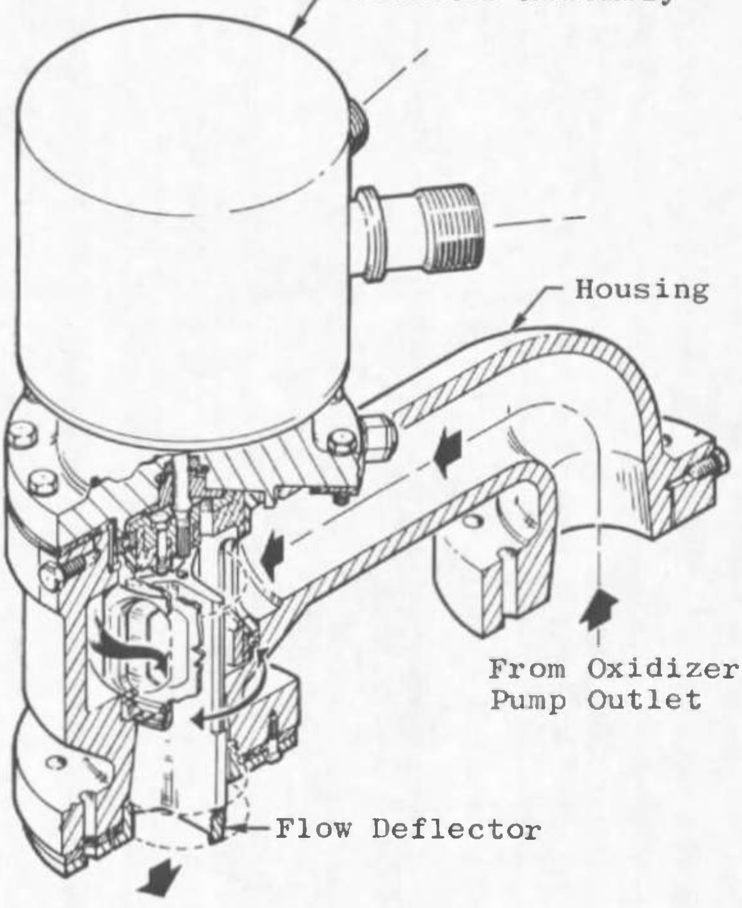


Fig. 7 Details of the PU Valve

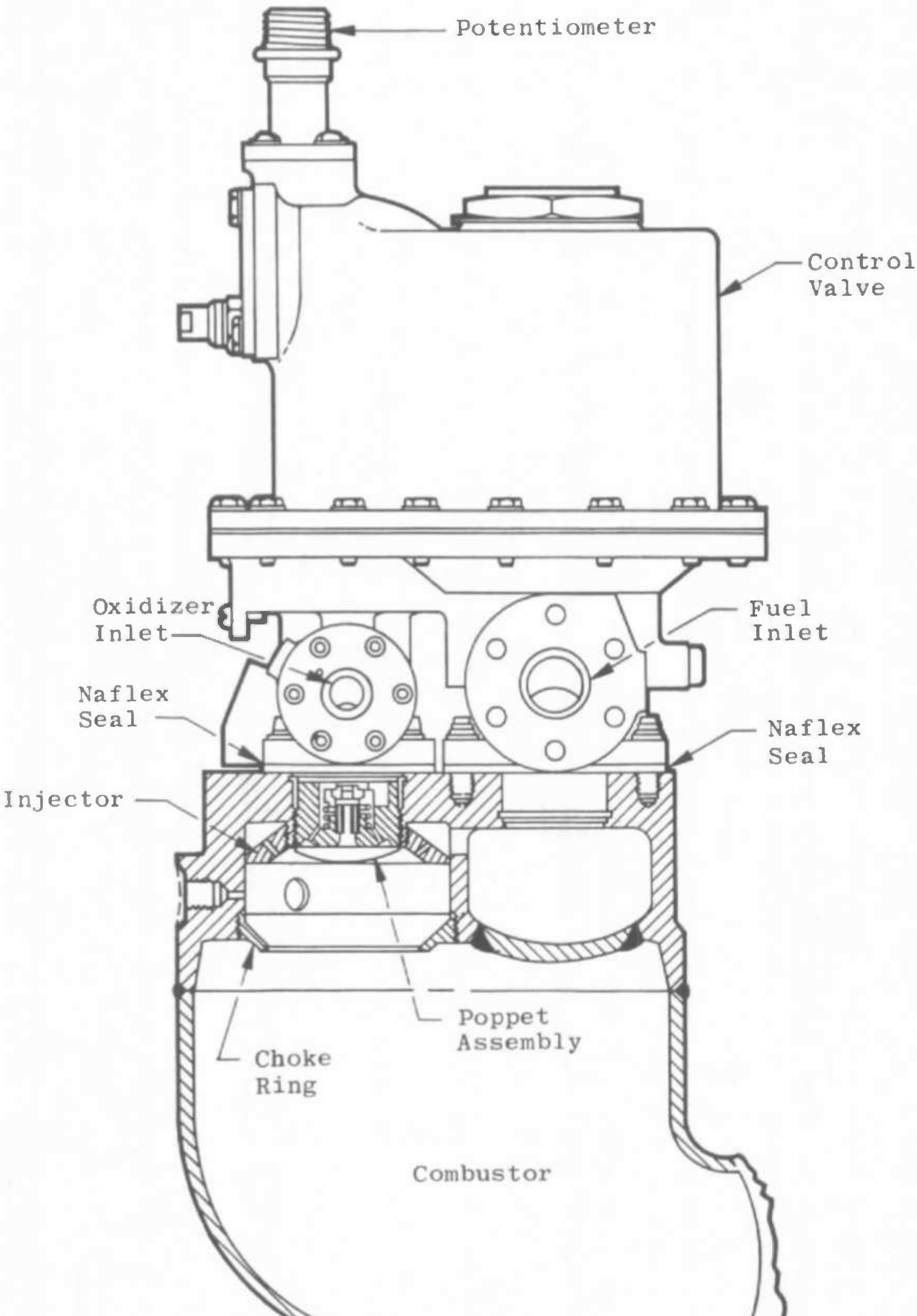


Fig. 8 Details of the GG Assembly

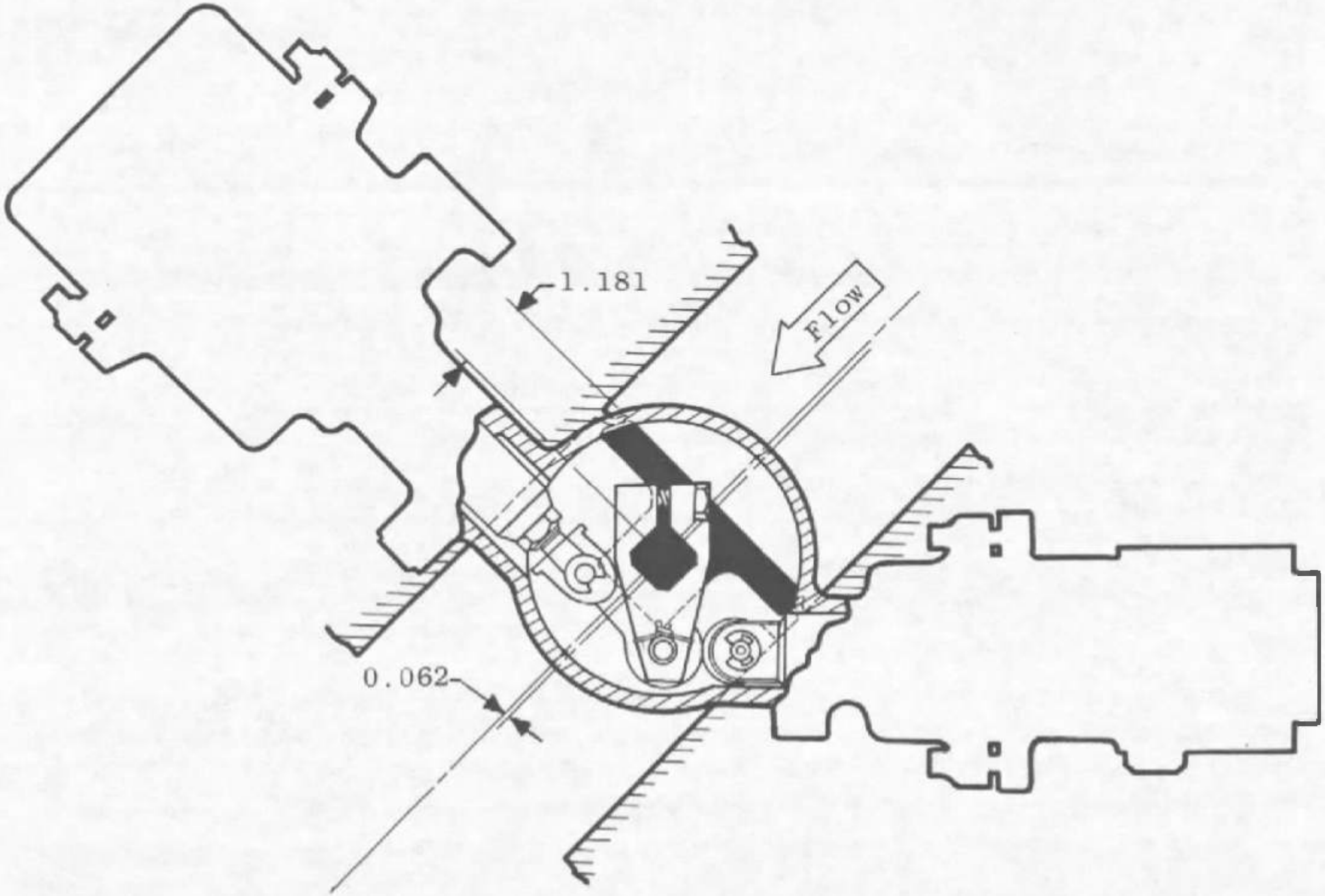


Fig. 9 Details of MOV

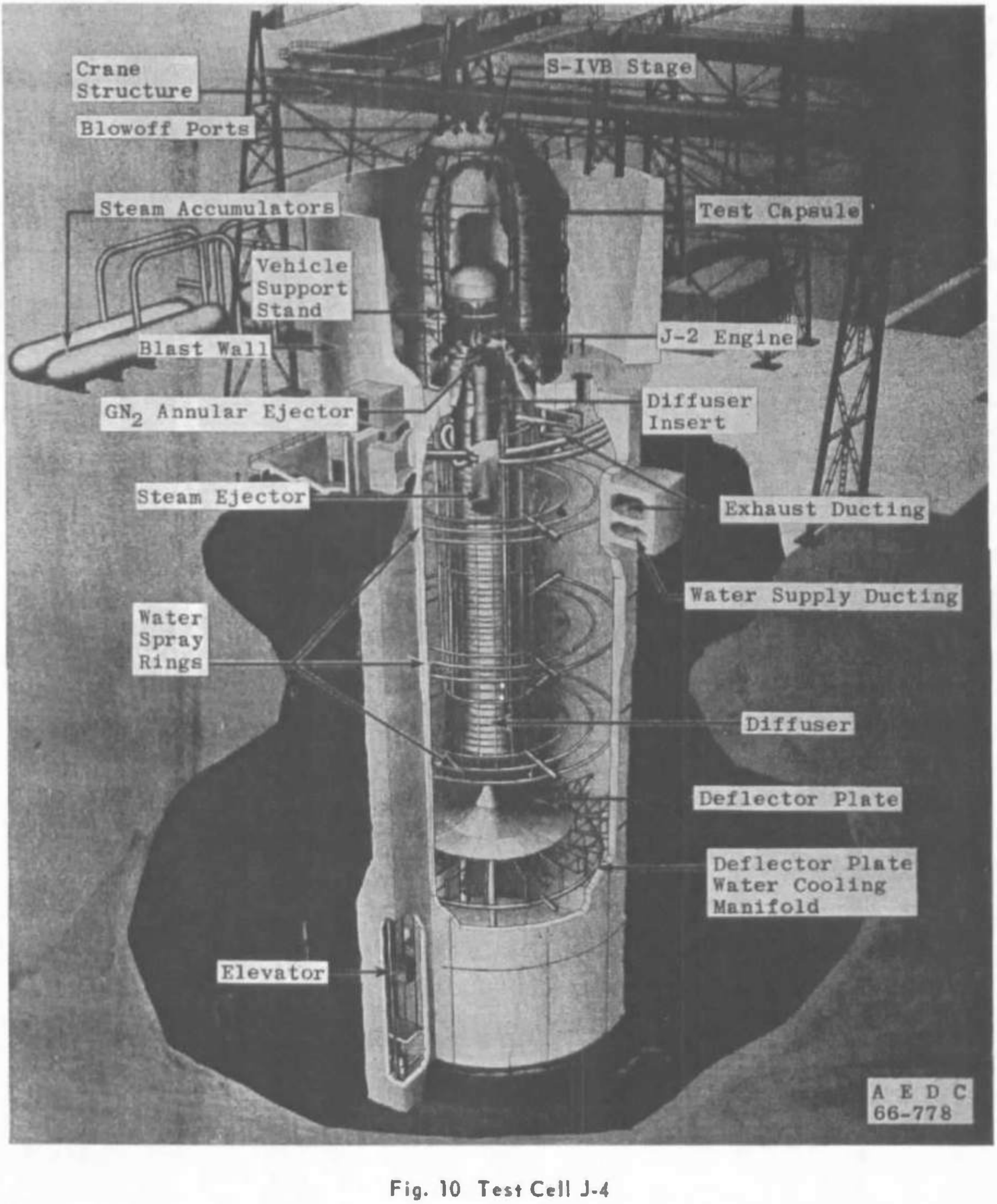
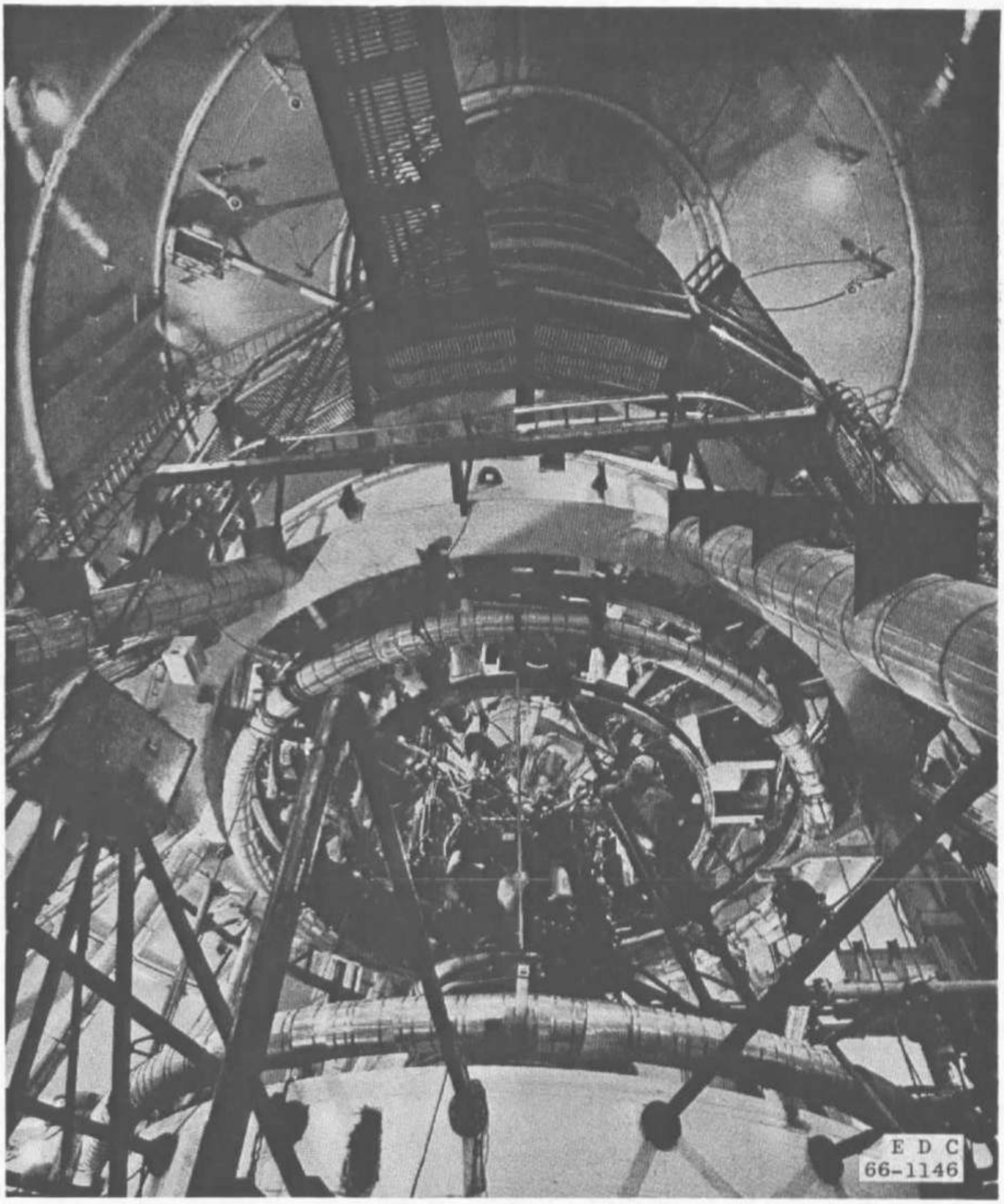
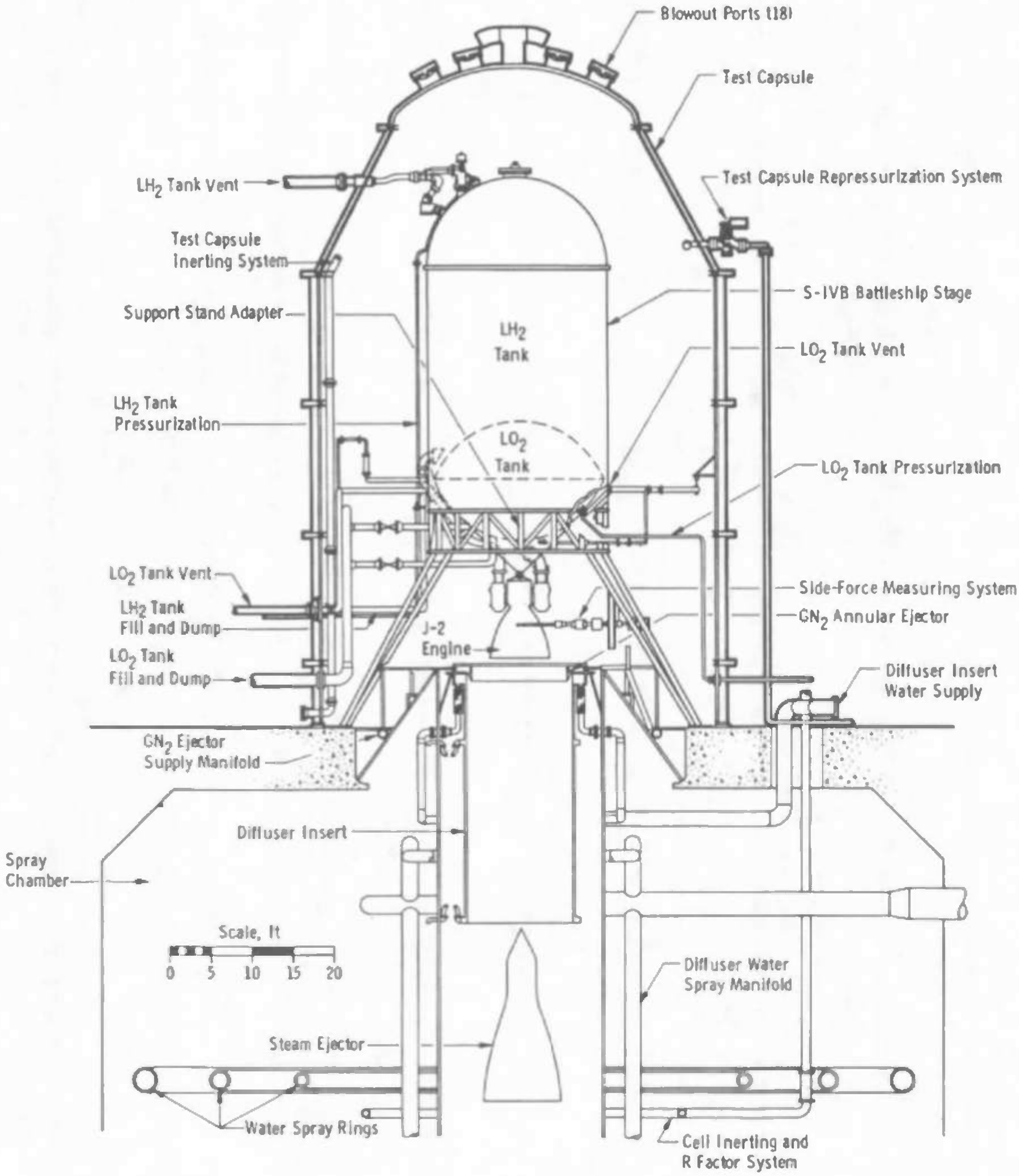


Fig. 10 Test Cell J-4



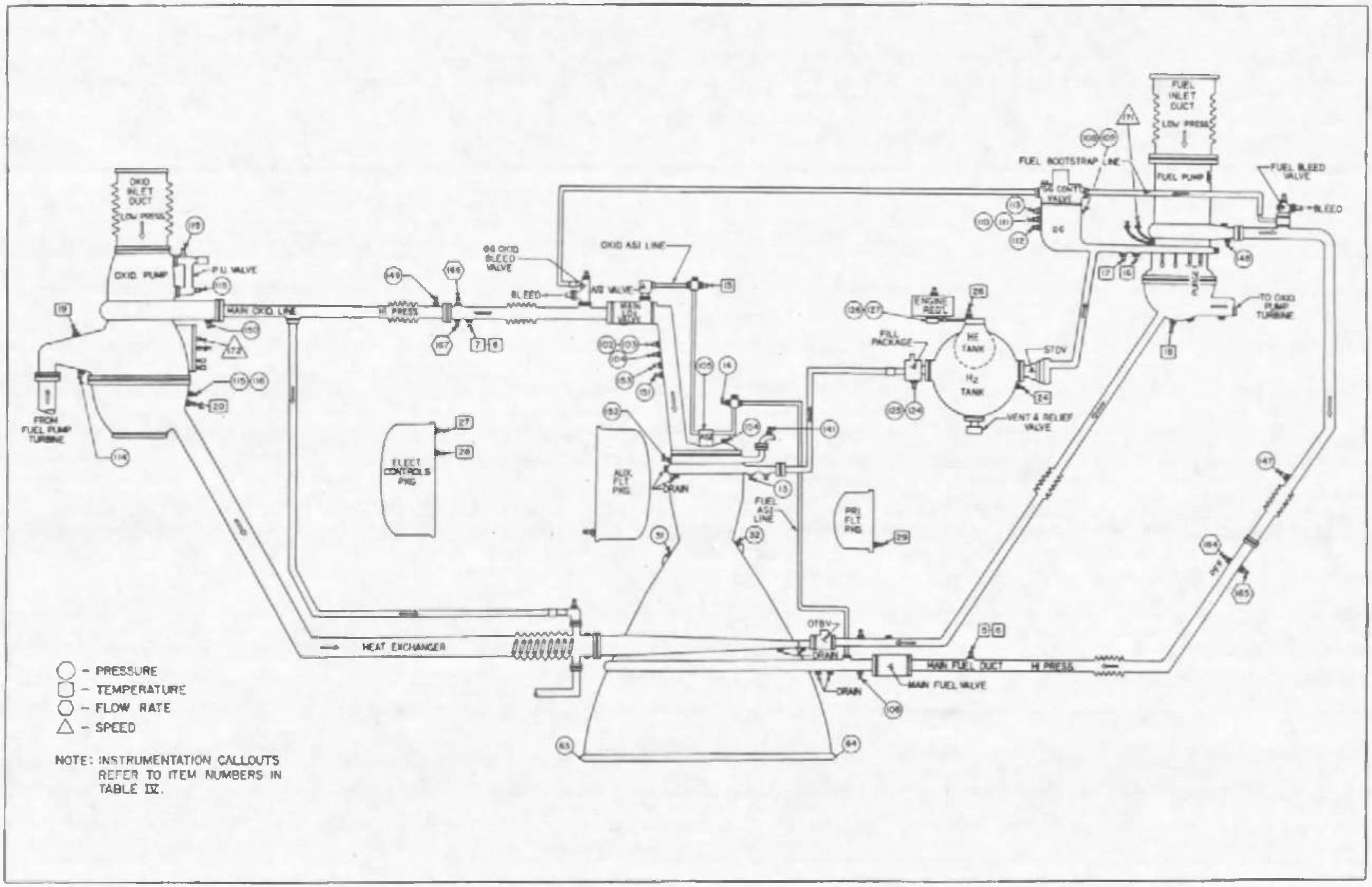
a. Photograph

Fig. 11 Test Article Installation in Test Cell J-4



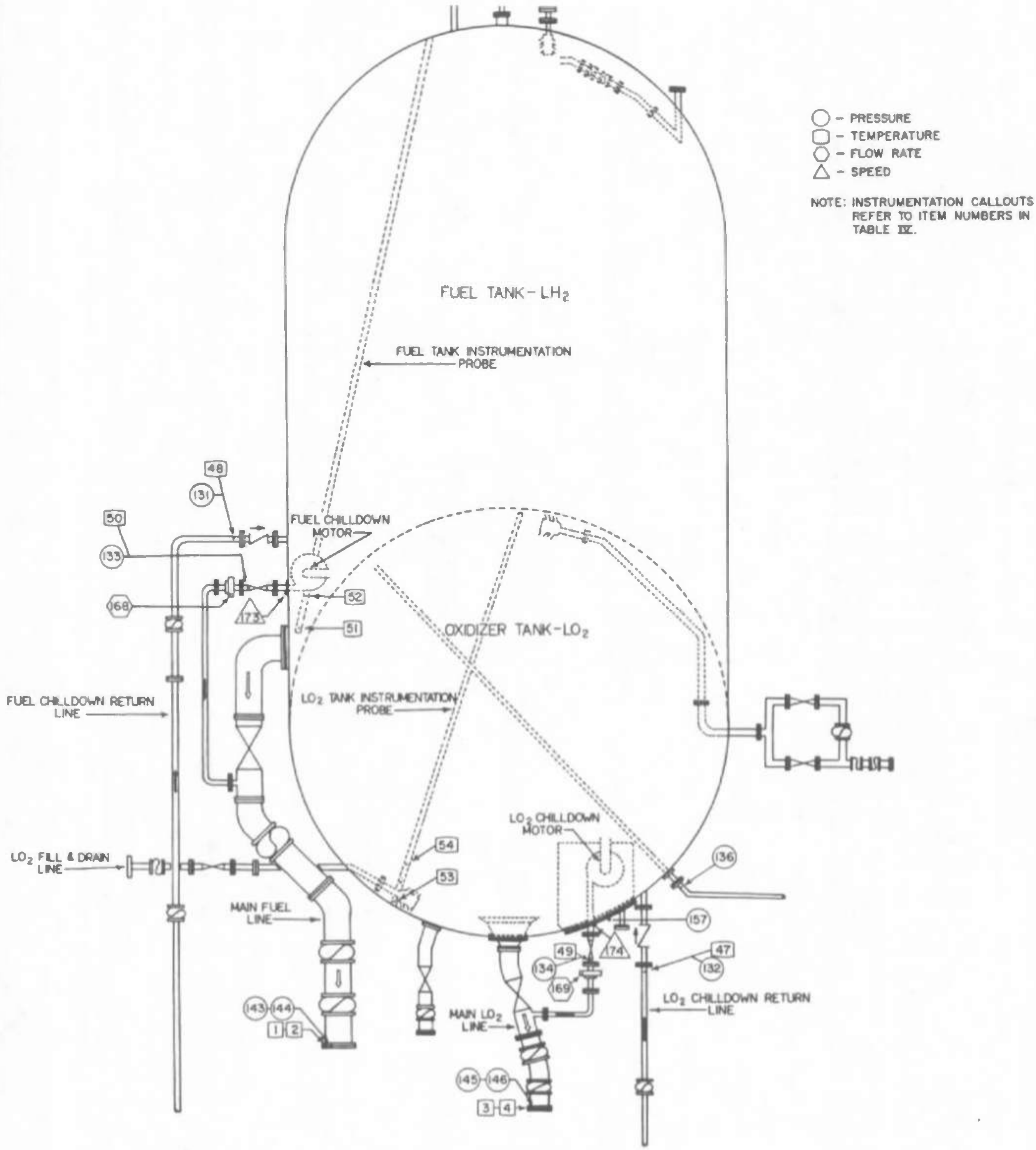
b. Elevation View

Fig. 11 Concluded

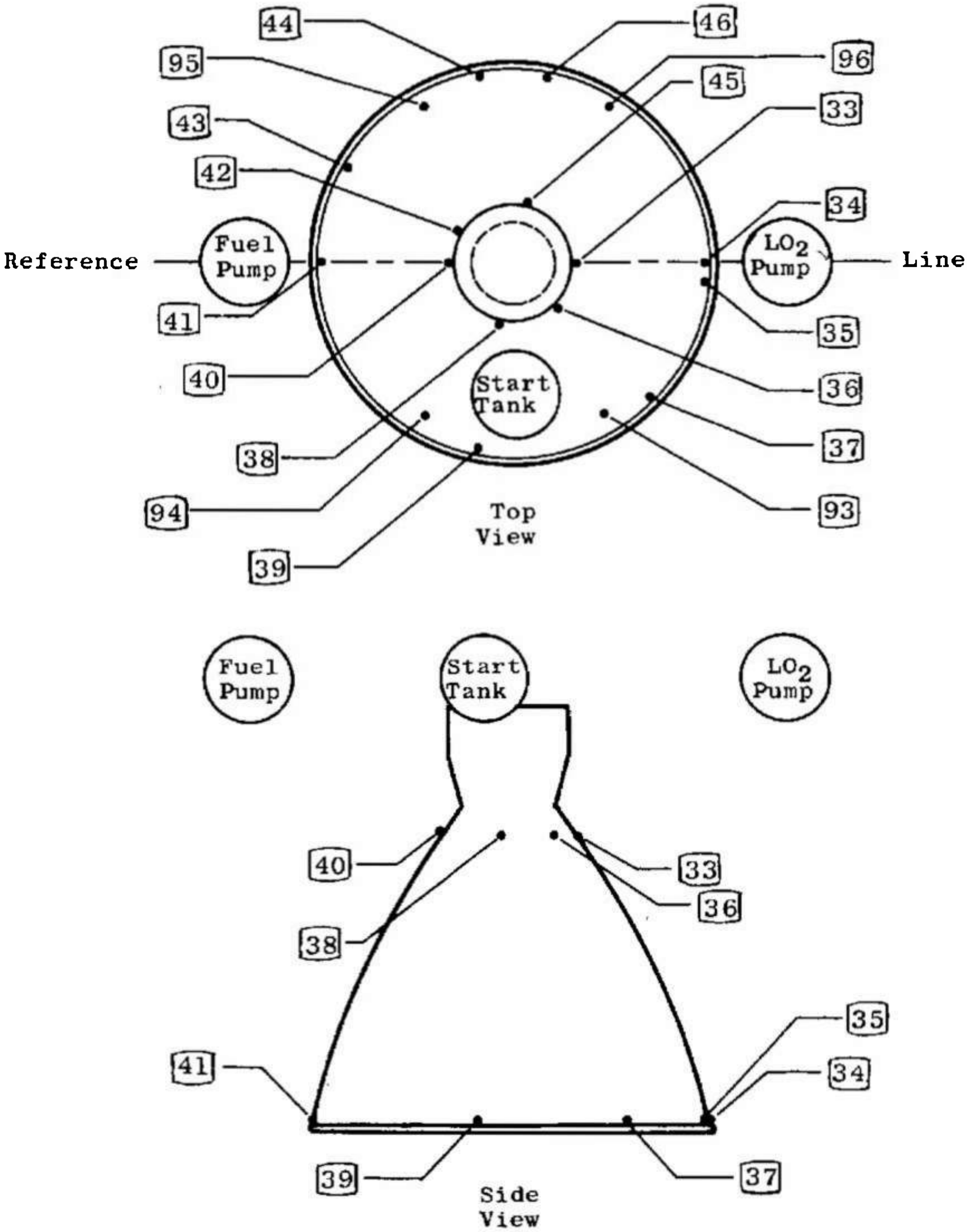


a. Engine Pressure Tap Locations

Fig. 12 Instrumentation of the J-2 Engine and S-IVB Stage



b. S-IVB Stage Instrumentation
Fig. 12 Continued



Note: Instrumentation callouts refer to item numbers in Table IV.

c. Thrust Chamber

Fig. 12 Concluded

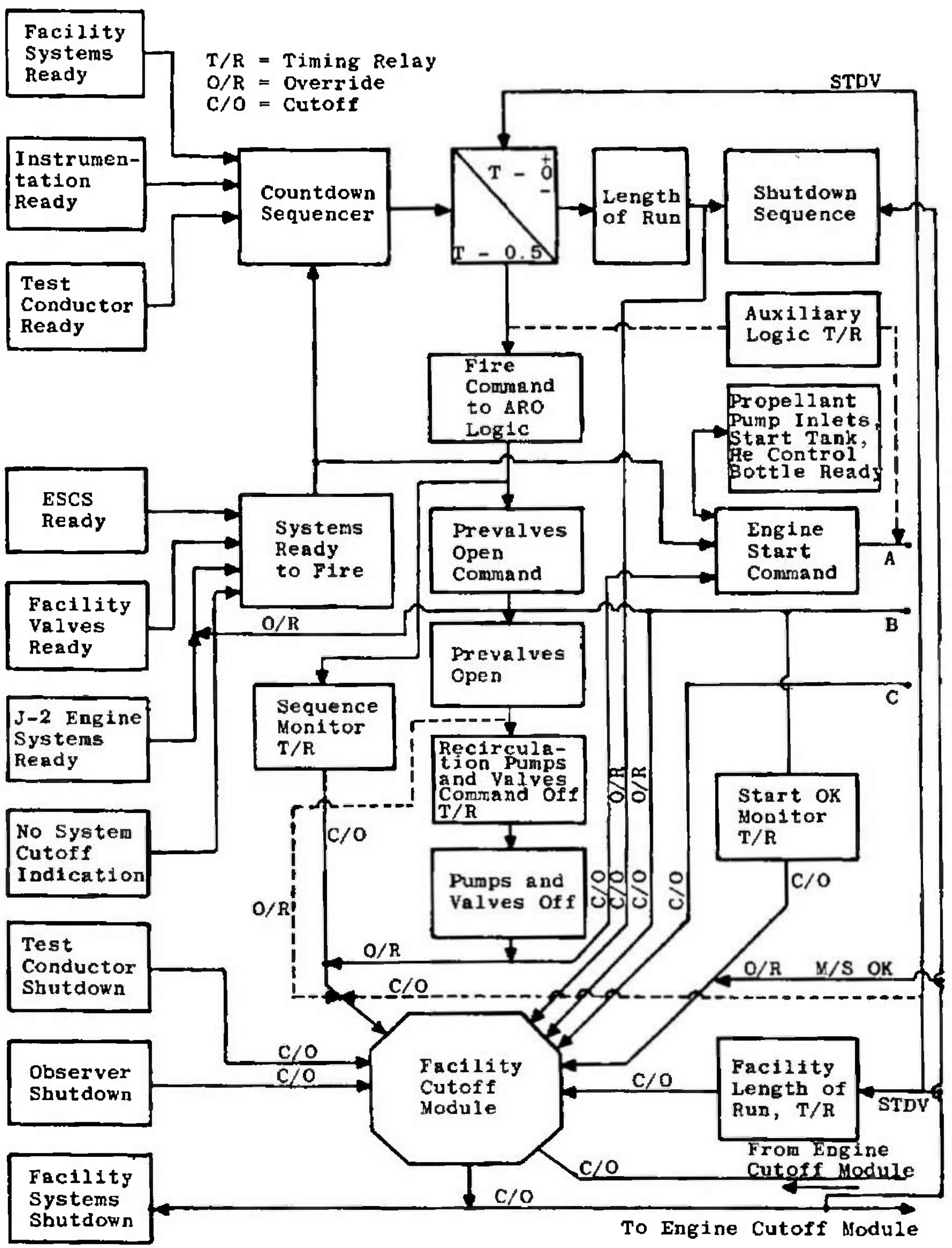


Fig. 13 Facility Logic Schematic

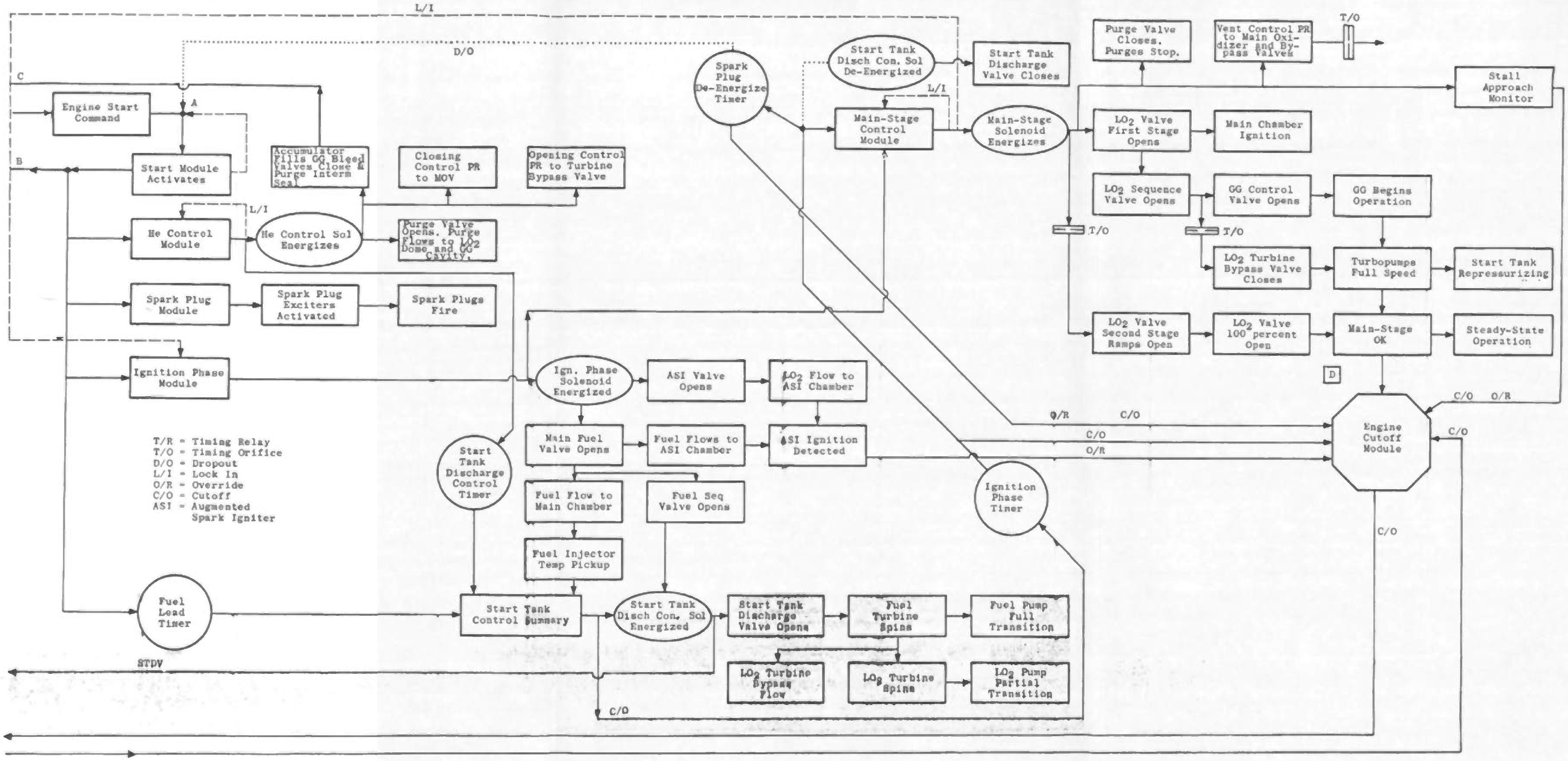


Fig. 14 Start Logic Schematic of the J-2 Engine

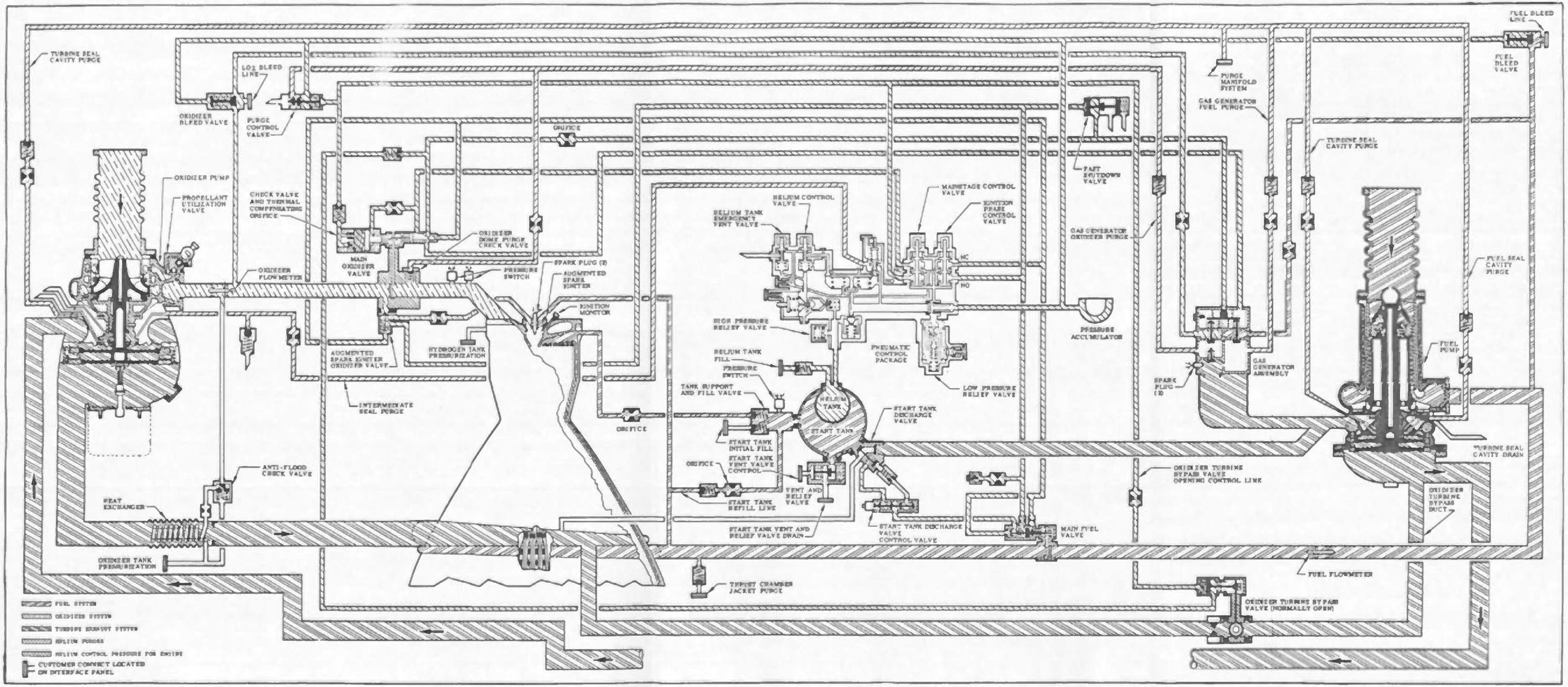


Fig. 15 Mechanical Schematic of the J-2 Engine

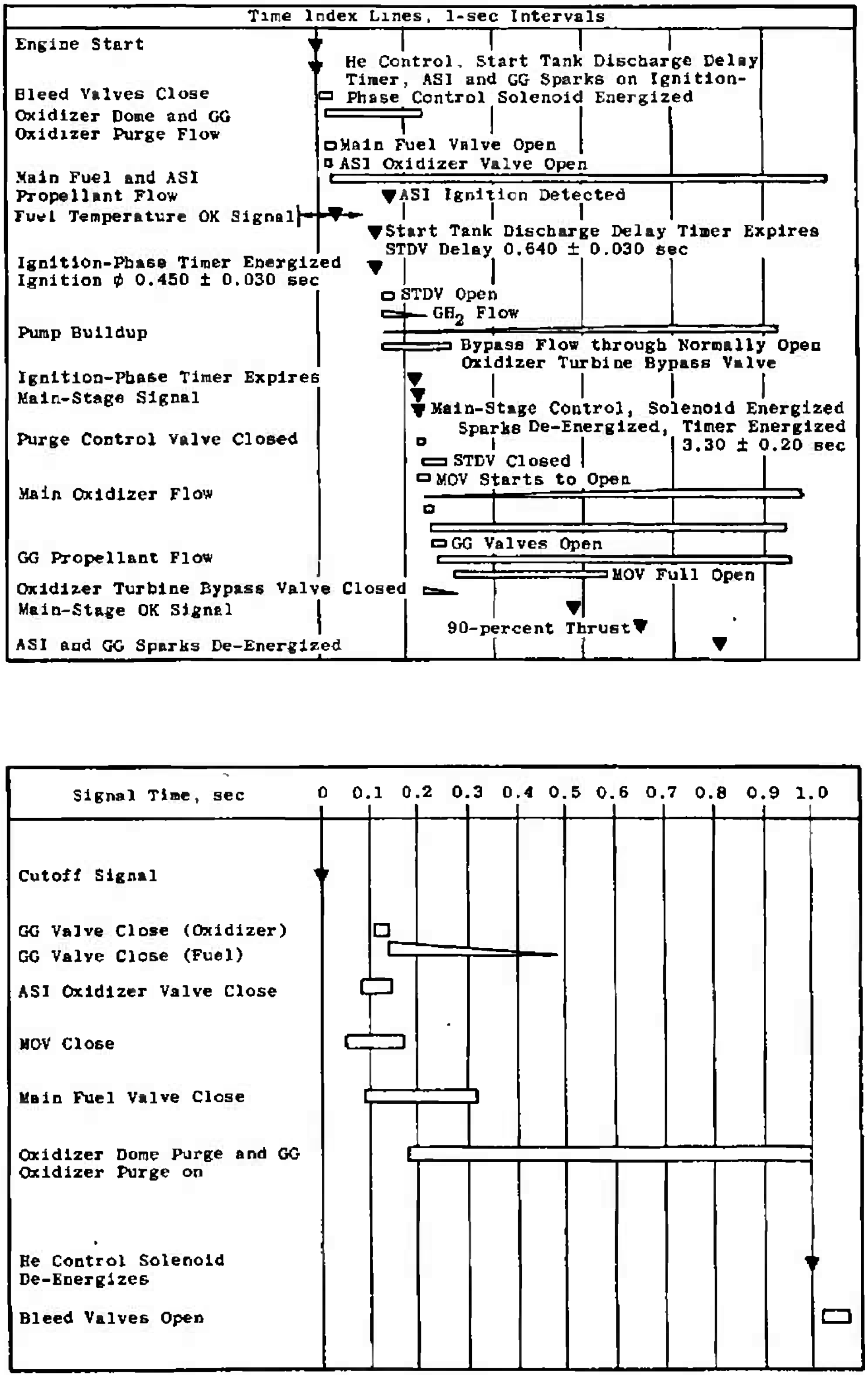


Fig. 16 Sequence Diagram of the J-2 Engine

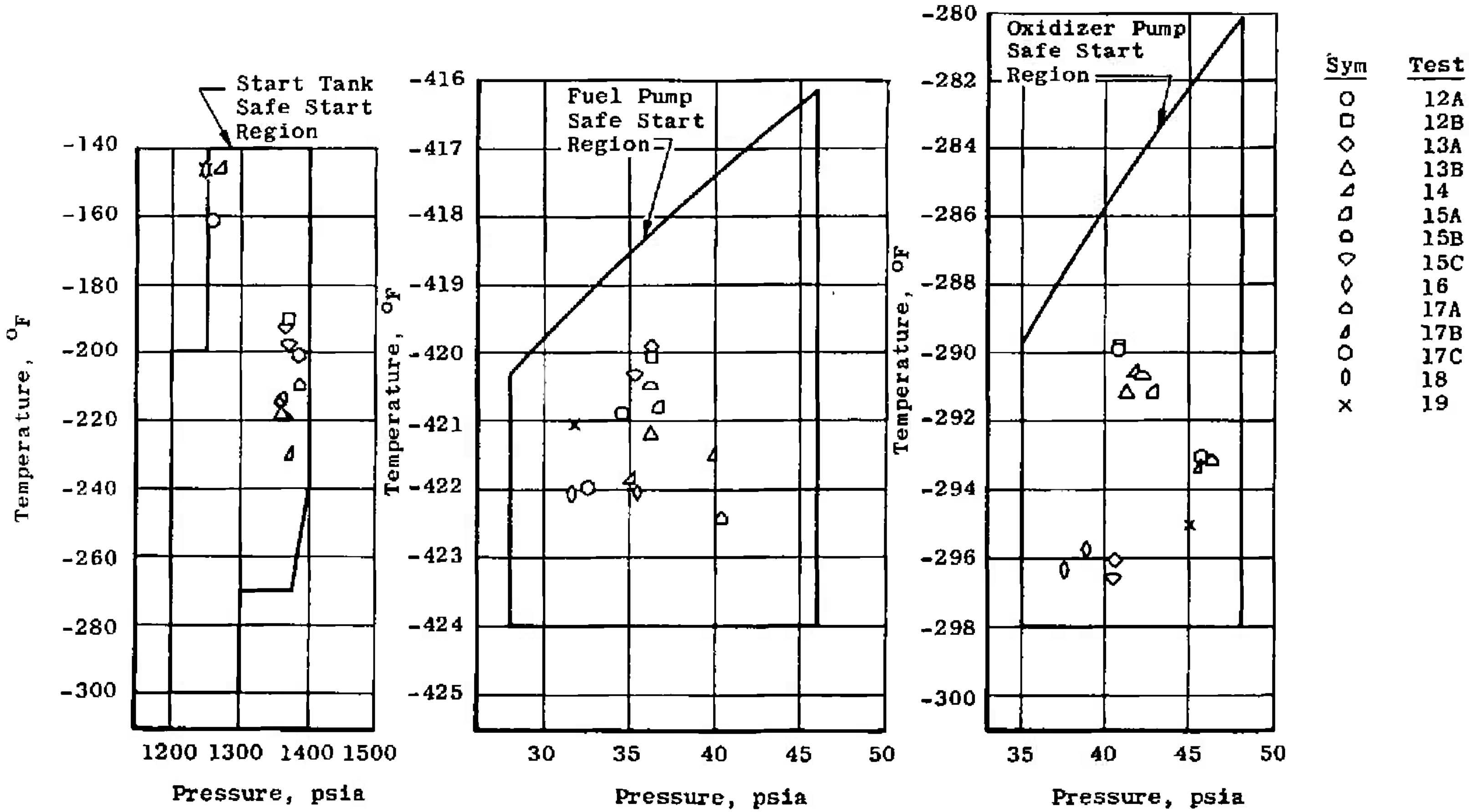


Fig. 17 Pump Inlet and Start Tank Conditions at ES

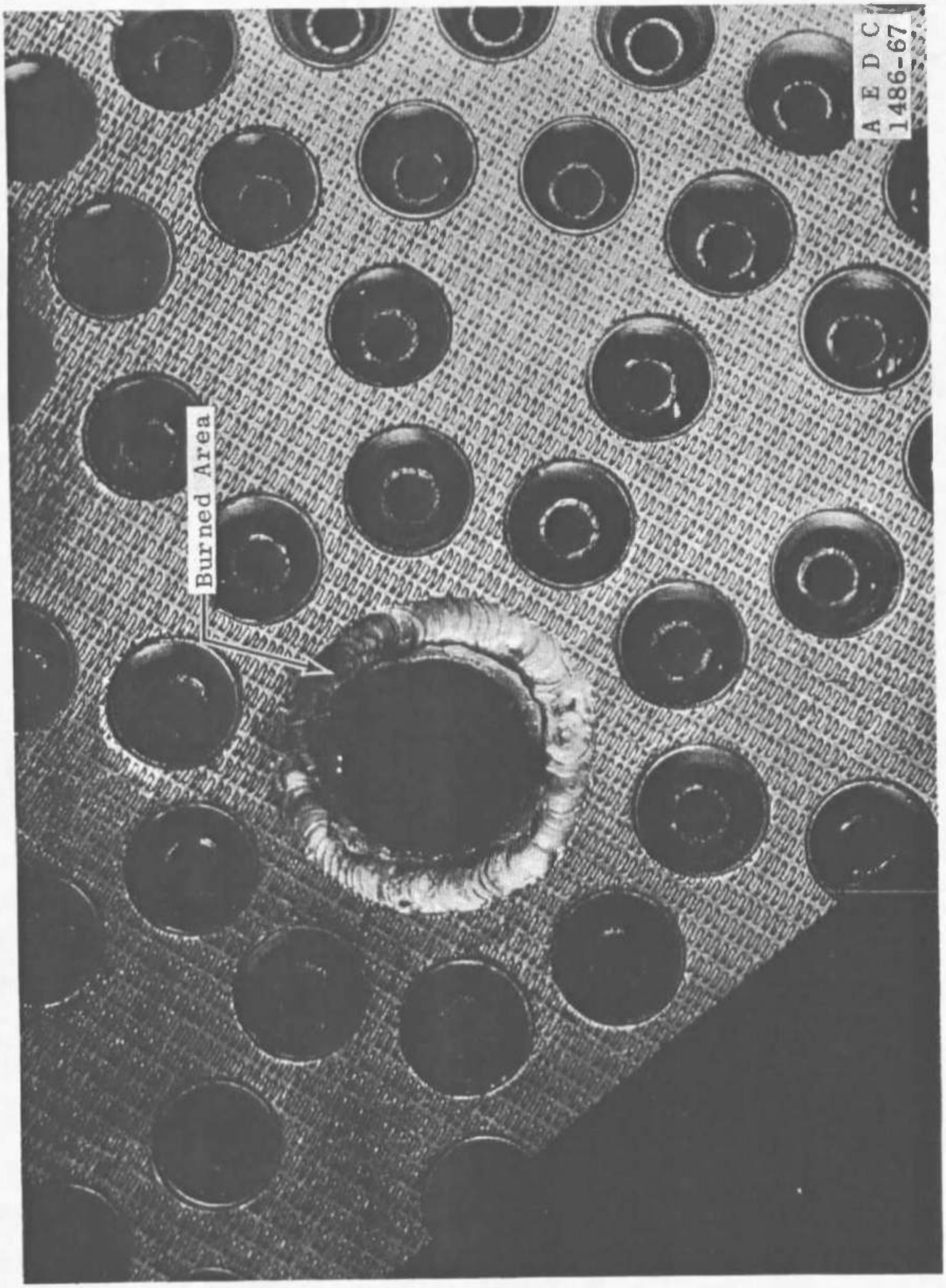
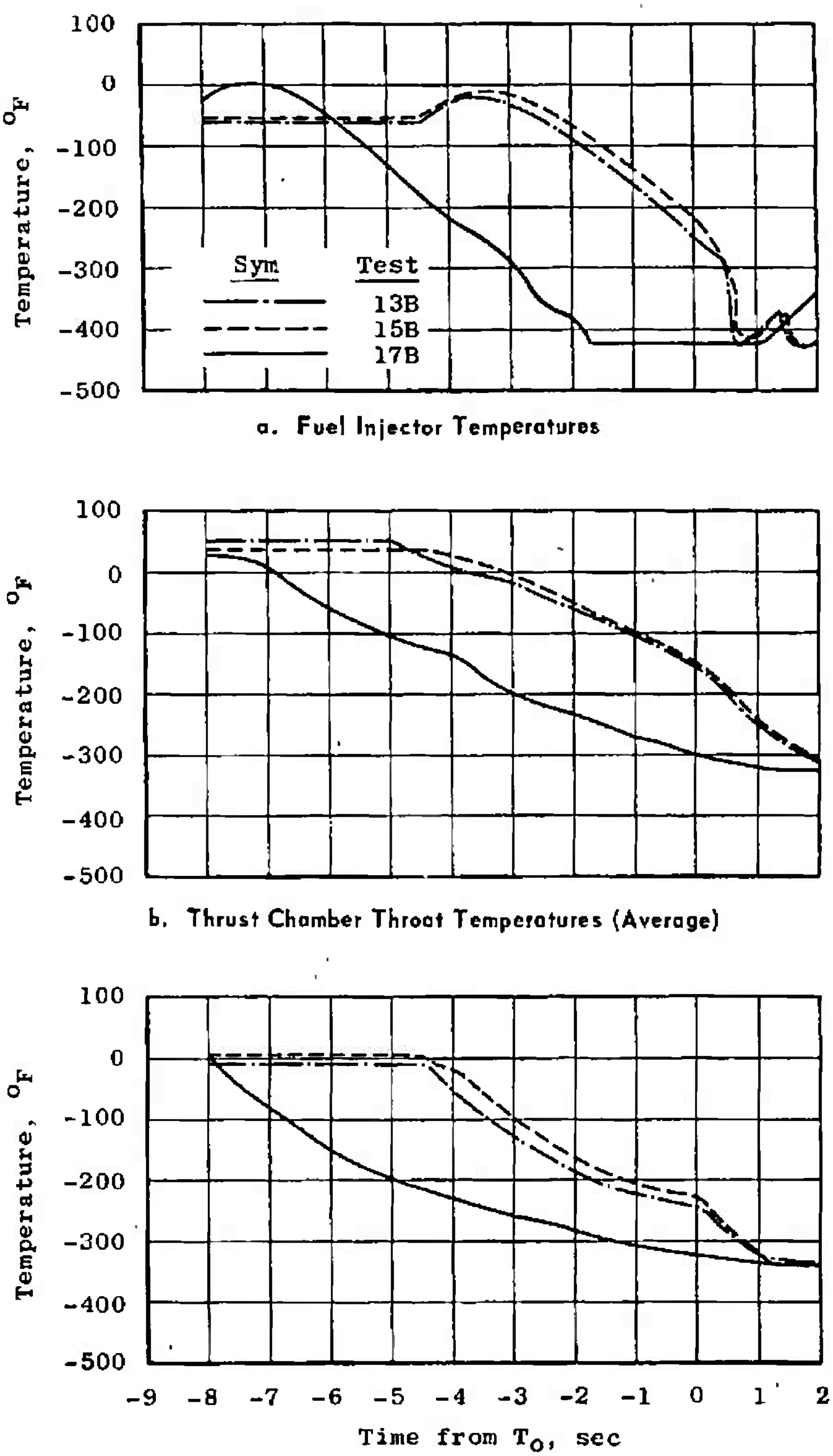


Fig. 18 Injector Face Damage, Post-Test 19



a. Fuel Injector Temperatures

b. Thrust Chamber Throat Temperatures (Average)

c. Thrust Chamber Exit Temperatures (Average)

Fig. 19 Thrust Chamber Temperatures during Fuel Lead

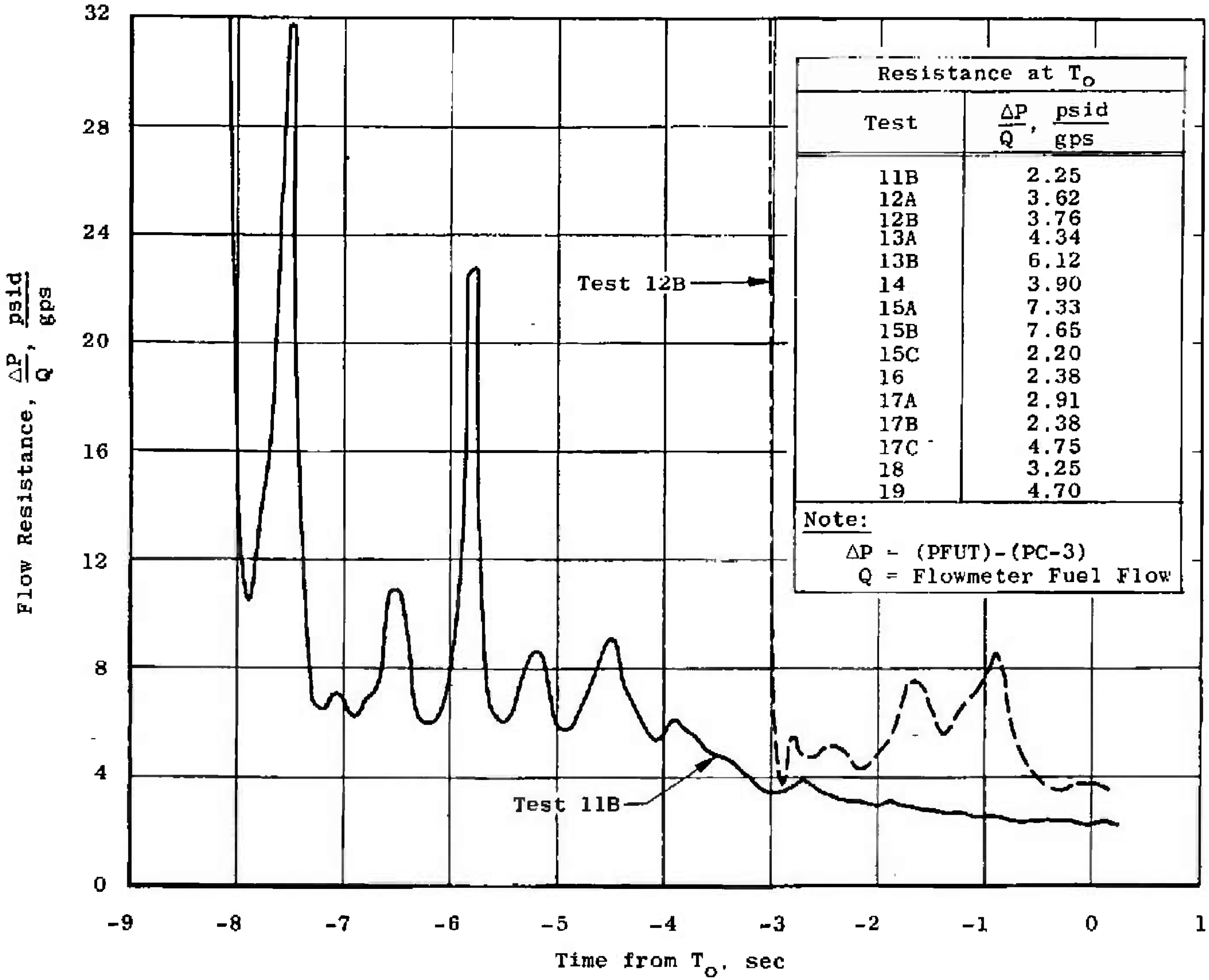


Fig. 20 Thrust Chamber Resistance to Fuel Flow during the Fuel Lead

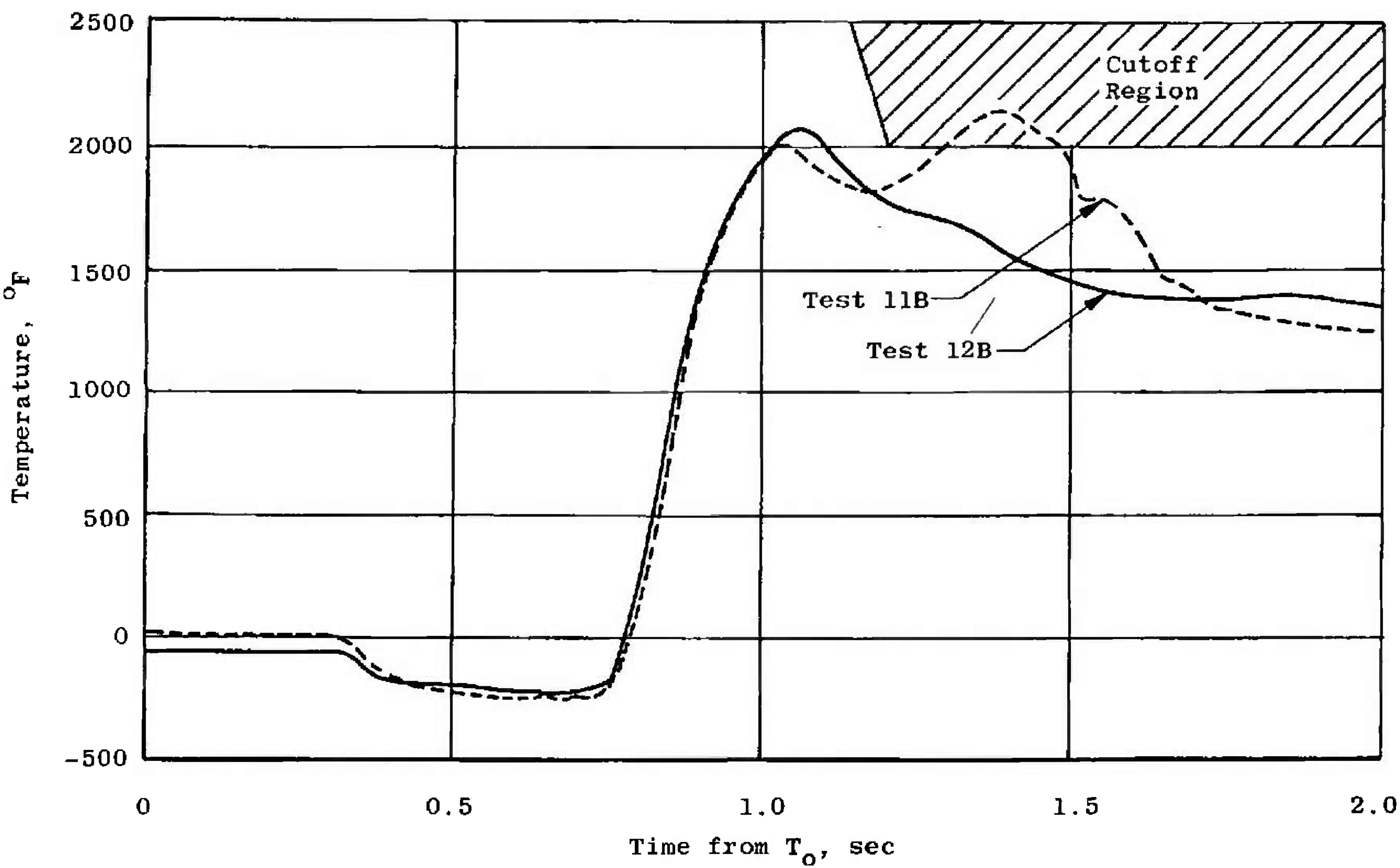
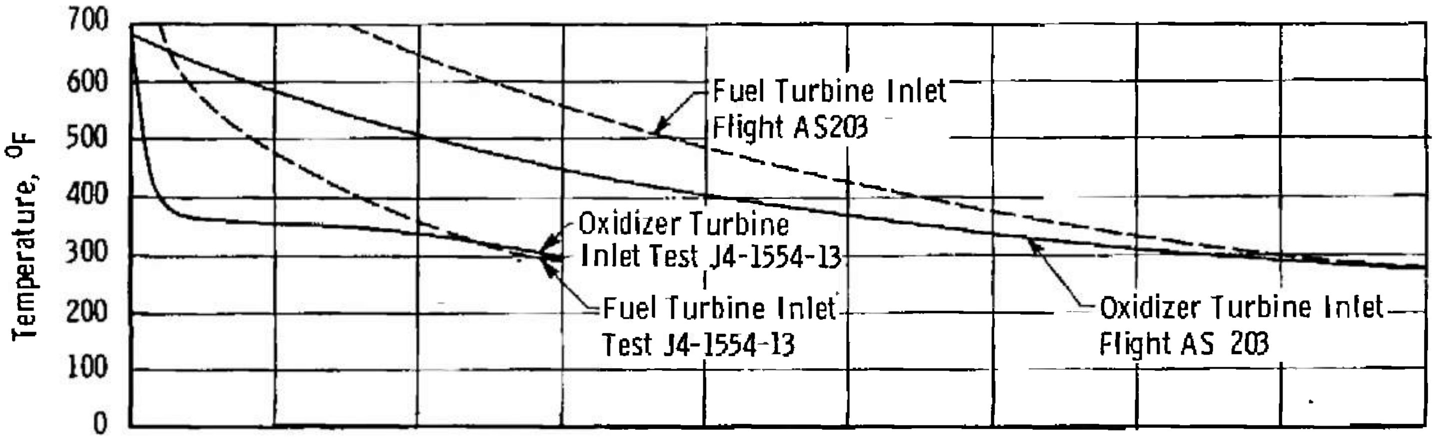
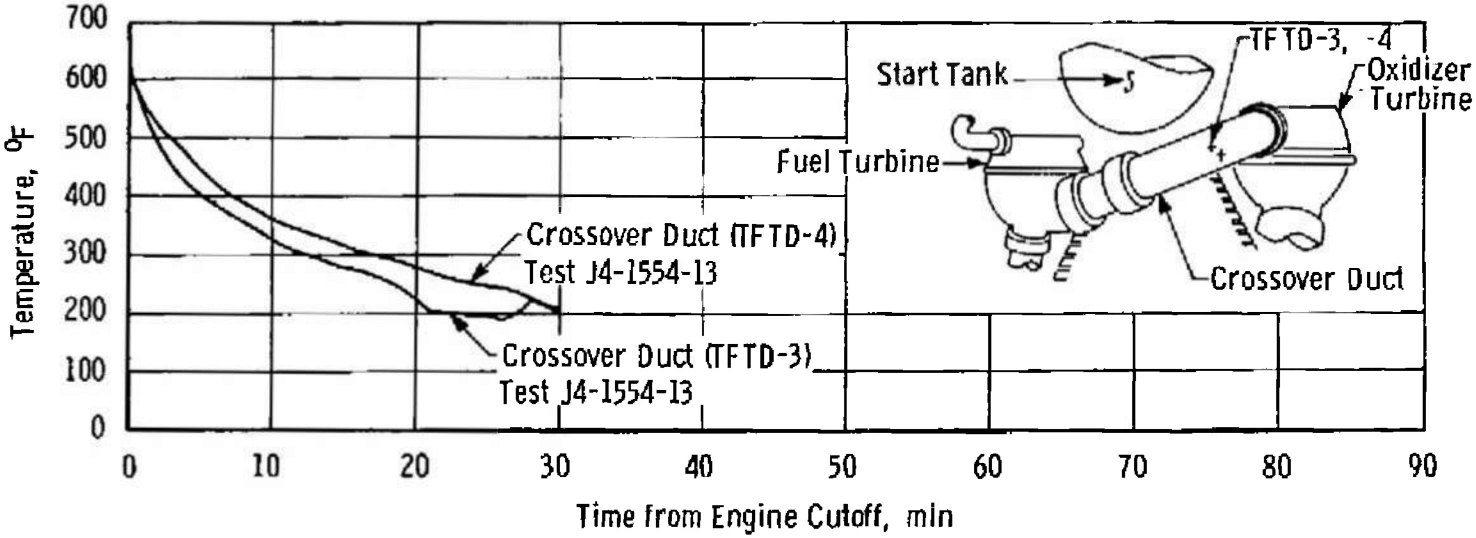


Fig. 21 Gas Generator Outlet Temperature, Tests 11B and 12B

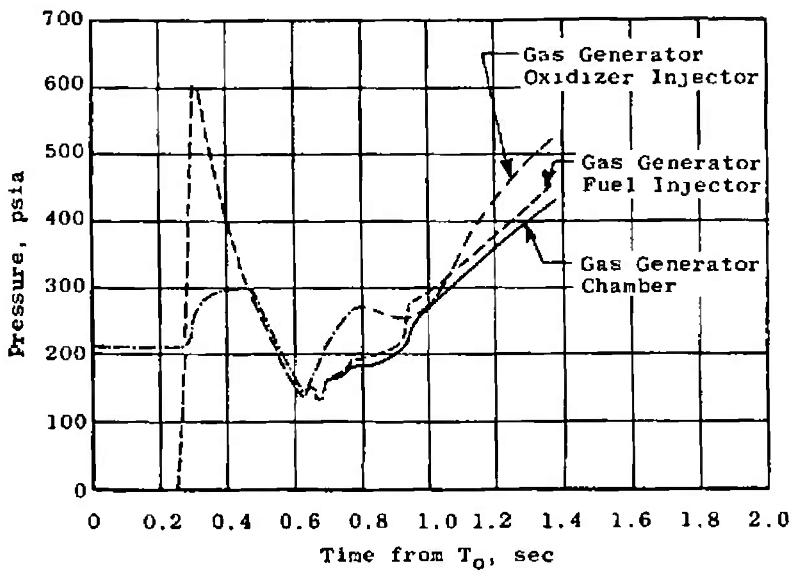


a. Turbine Inlet Temperatures

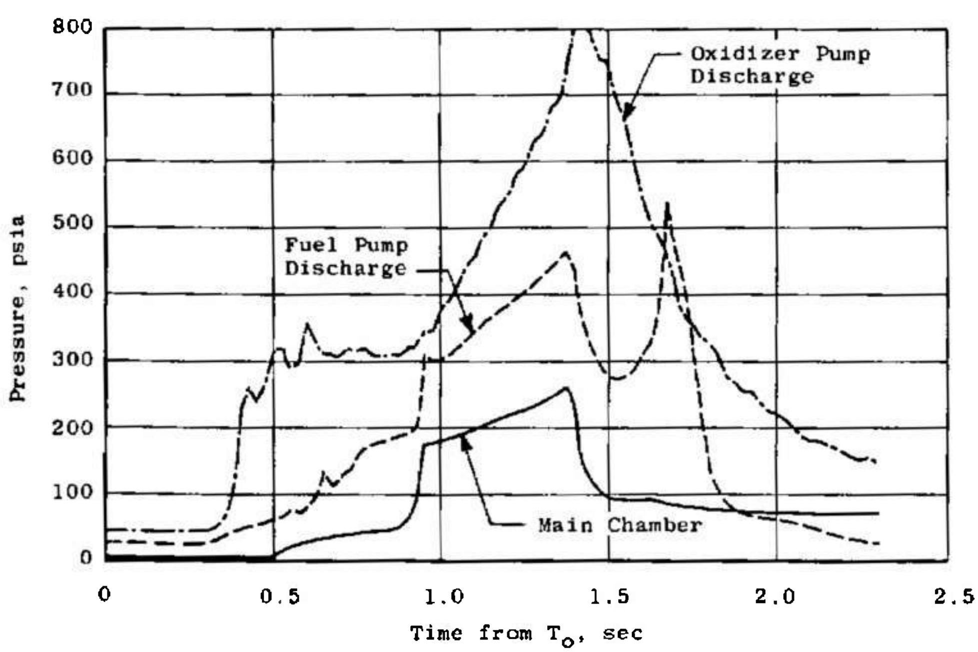


b. Crossover Duct Skin Temperatures

Fig. 22 Turbine Hardware Temperatures, AS 203 and Test 138

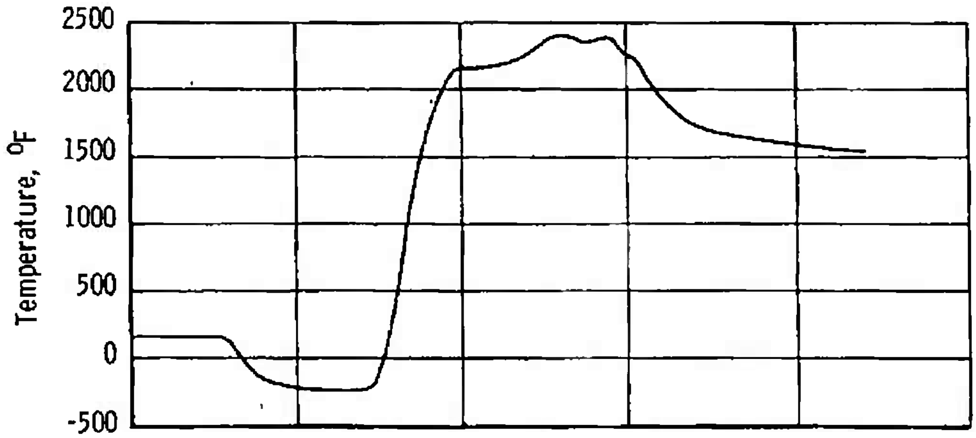


a. Gas Generator Transient Pressures

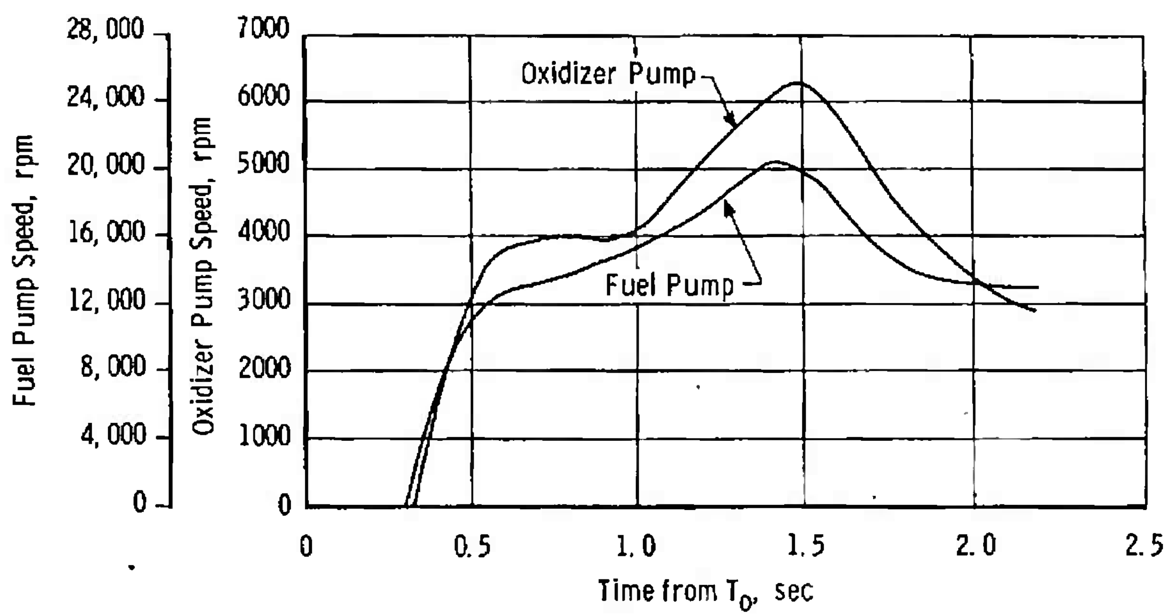


b. Propellant Systems Transient Pressures

Fig. 23 Test 13B Engine Start Transient



c. Gas Generator Outlet Temperature



d. Propellant Pump Speeds

Fig. 23 Concluded

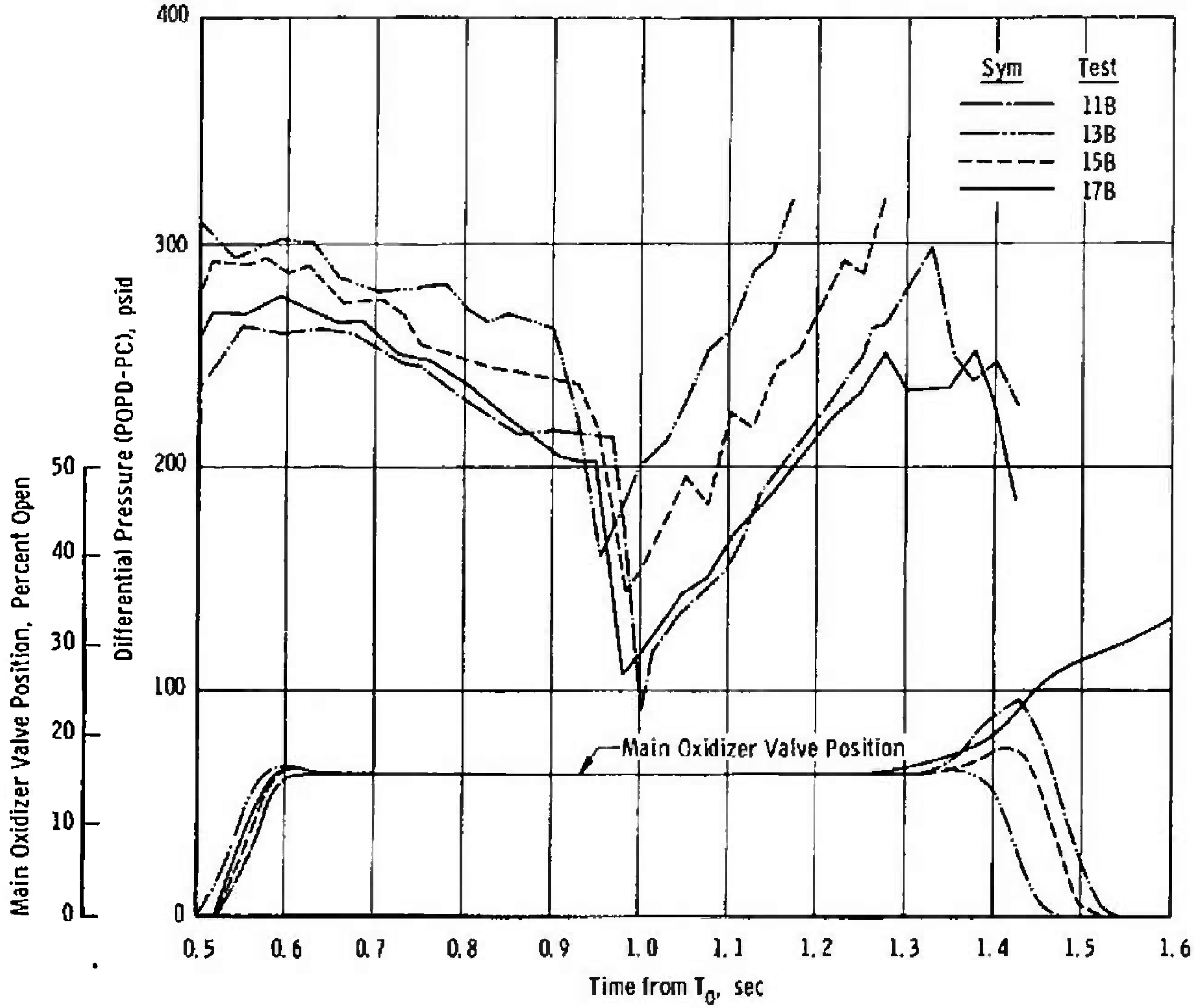
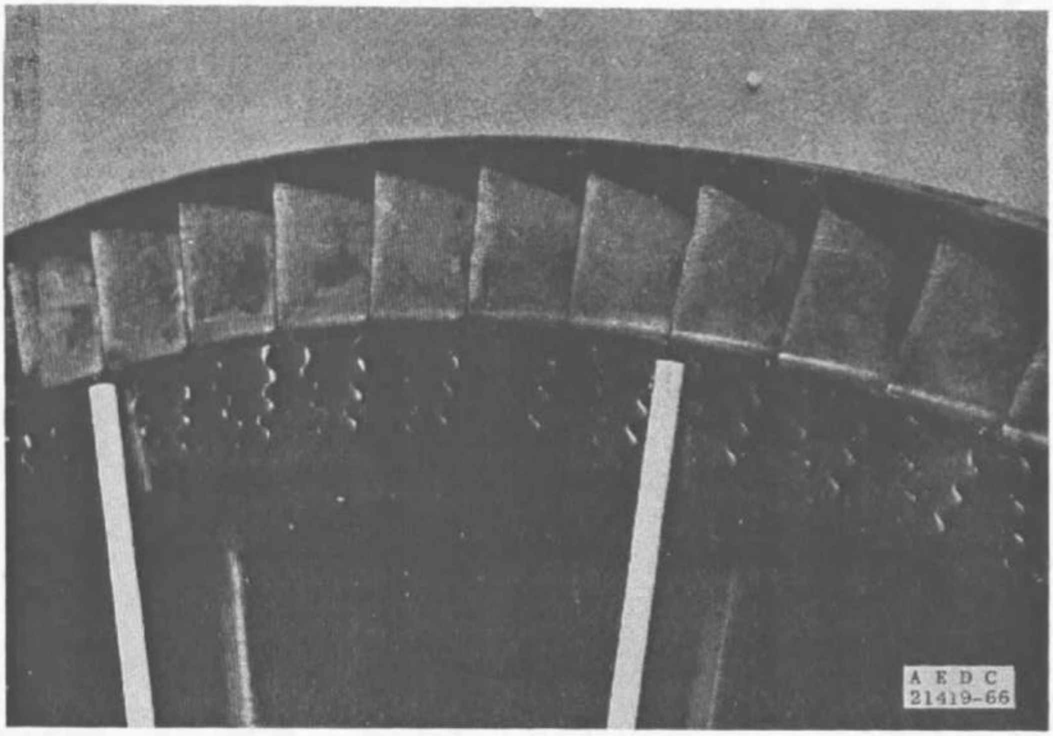
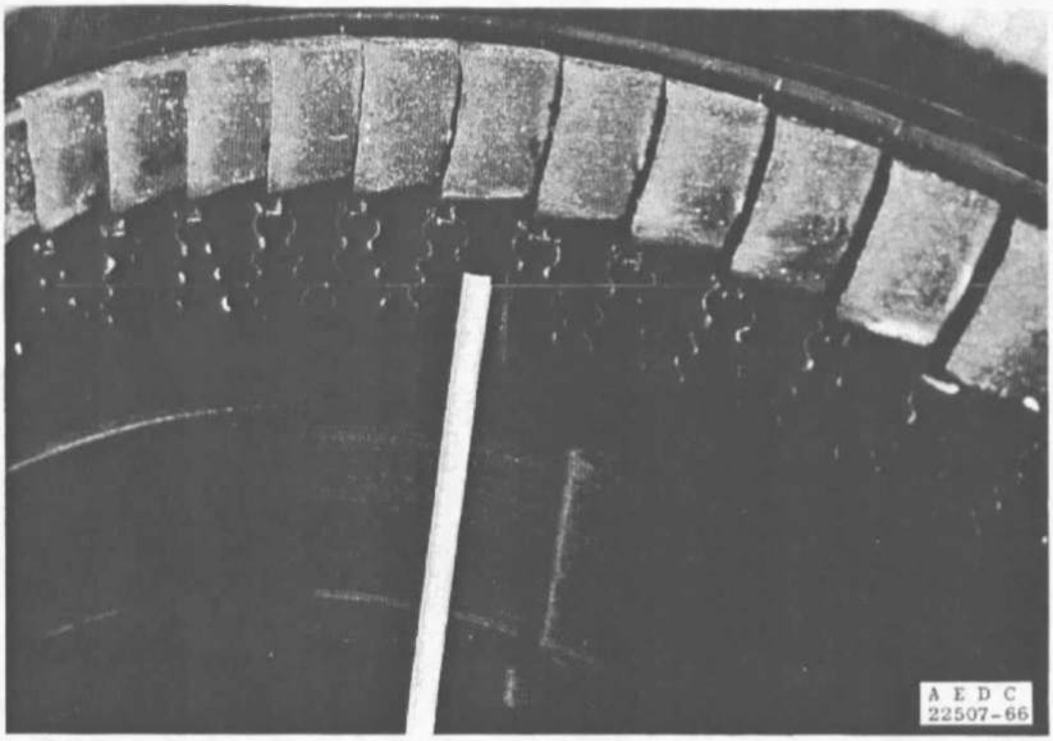


Fig. 24 Start Transient Differential Pressures across MOV and Valve Position

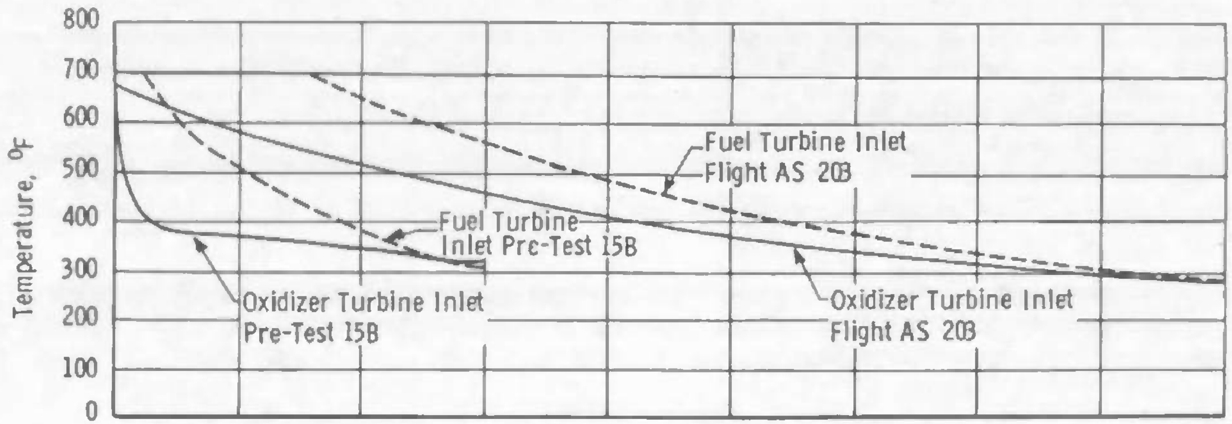


a. Post-Test 11

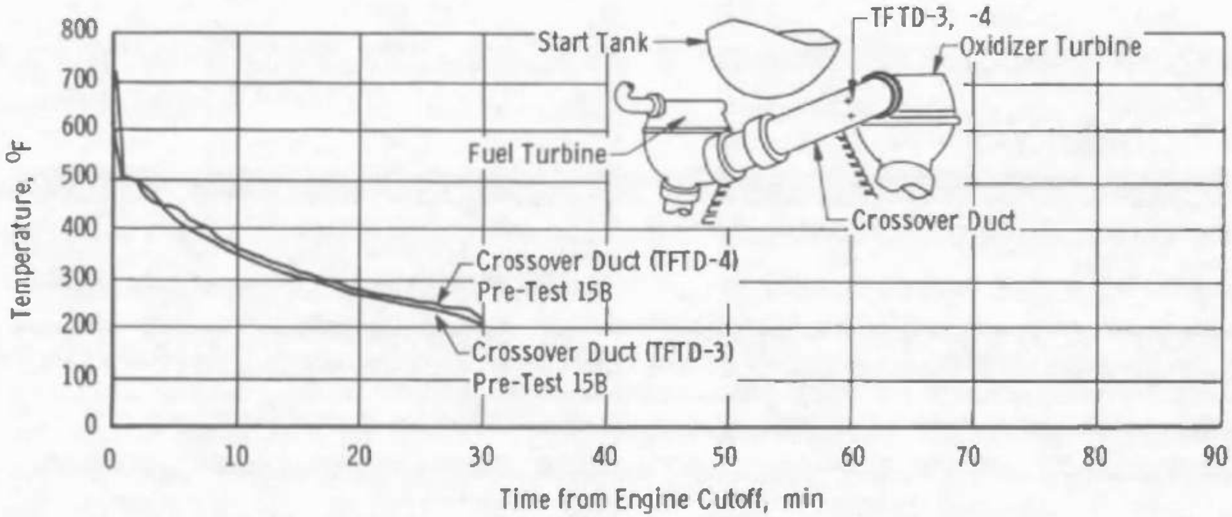


b. Post-Test 13

Fig. 25 First-Stage Fuel Turbine Wheel Erosion



a. Turbine Inlet Temperatures



b. Crossover Duct Skin Temperatures

Fig. 26 Turbine Hardware Temperatures, AS 203 and Test 15B

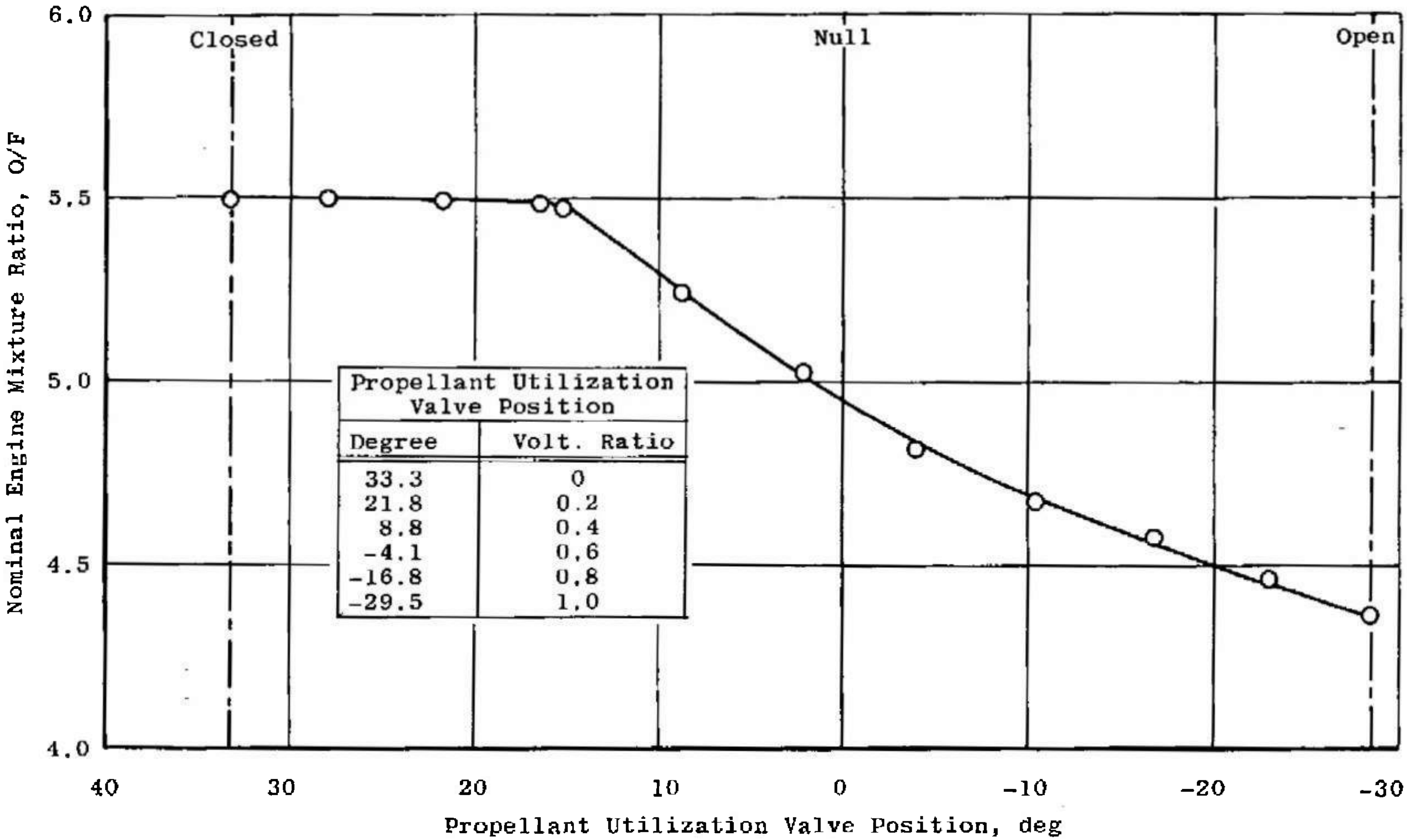


Fig. 27 Nominal Mixture Ratio of the J-2 Engine

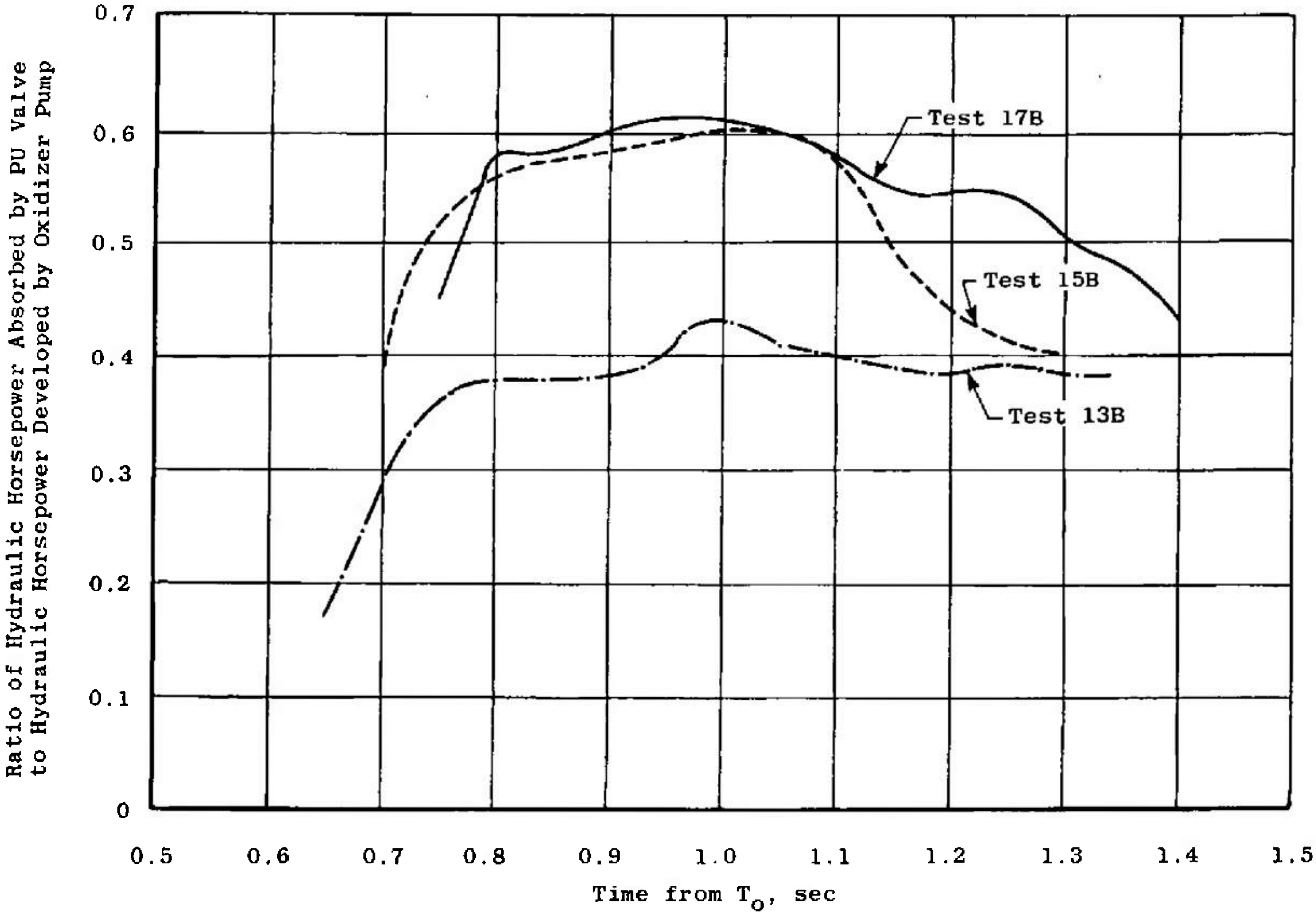
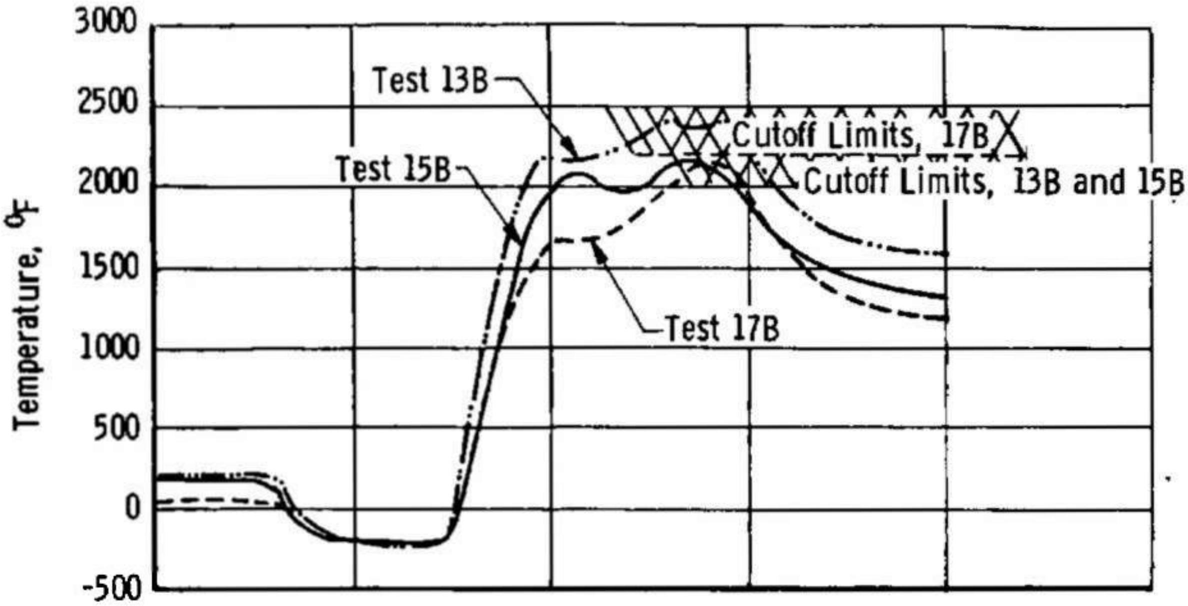
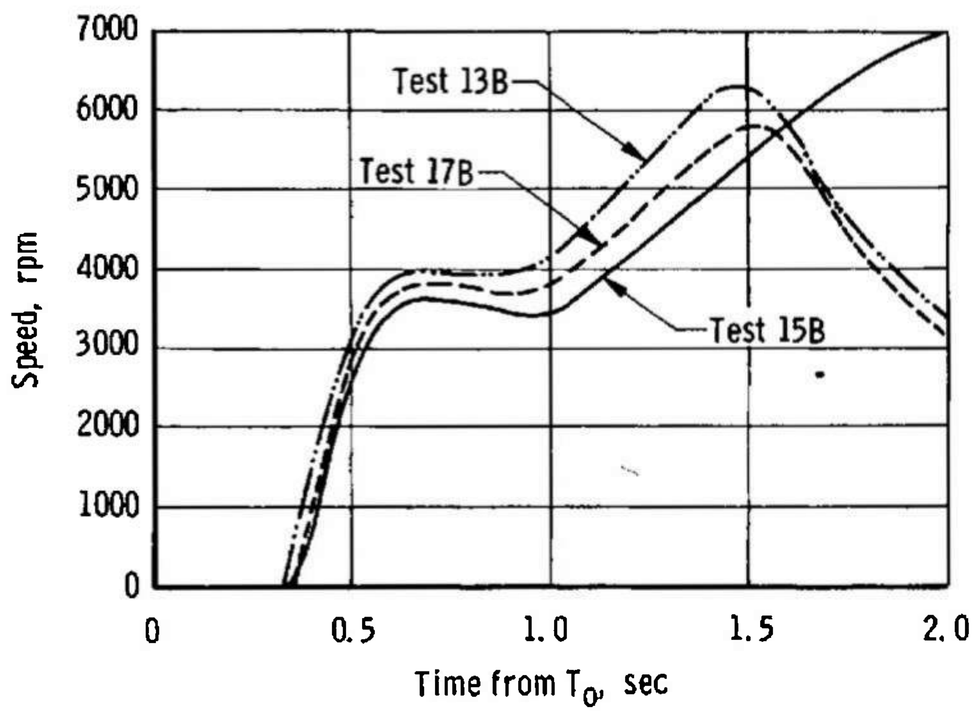


Fig. 28 Hydraulic Horsepower Absorbed by PU Valve

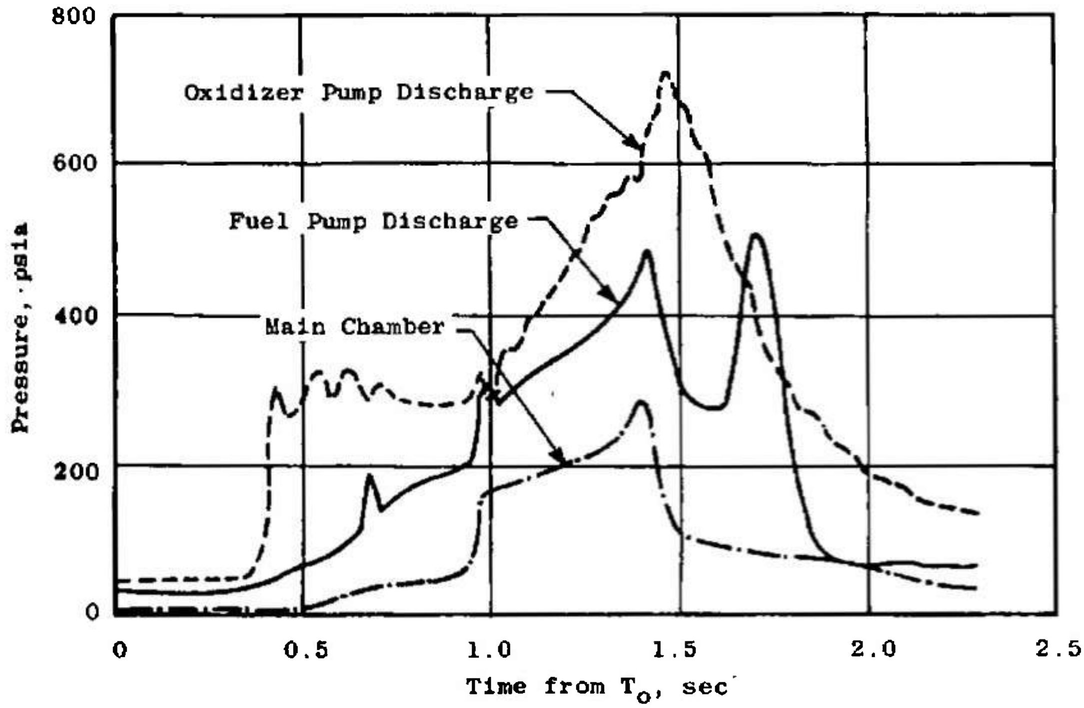


a. Gas Generator Outlet Temperature

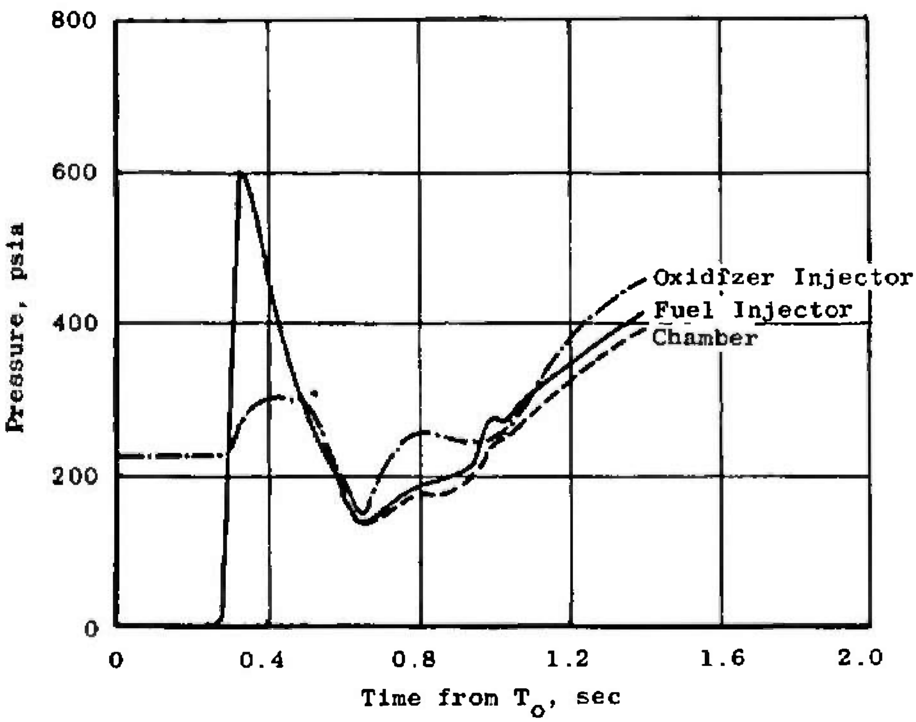


b. Oxidizer Pump Speed

Fig. 29 Oxidizer Pump Spin Speed and GGOT, Re-Start Tests

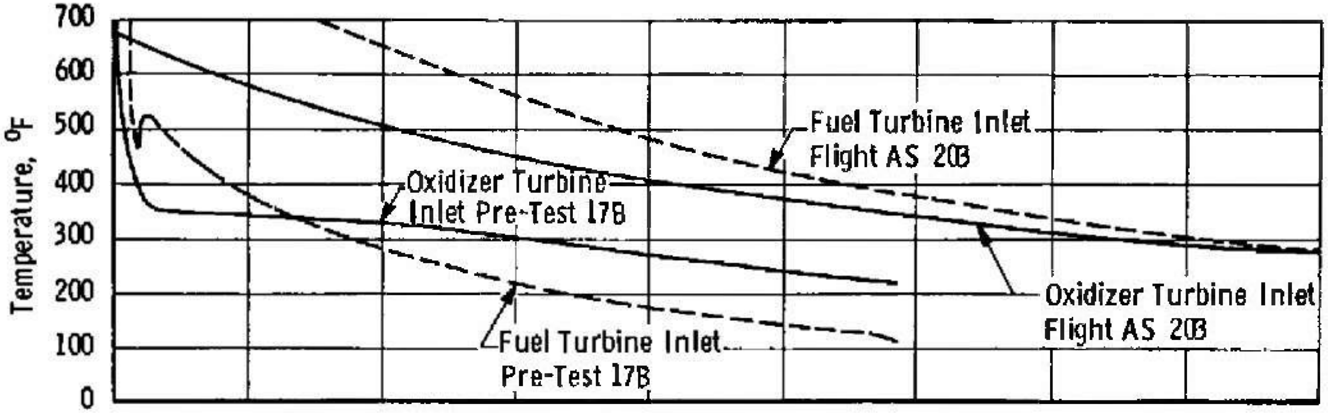


a. Propellant System Transient Pressures

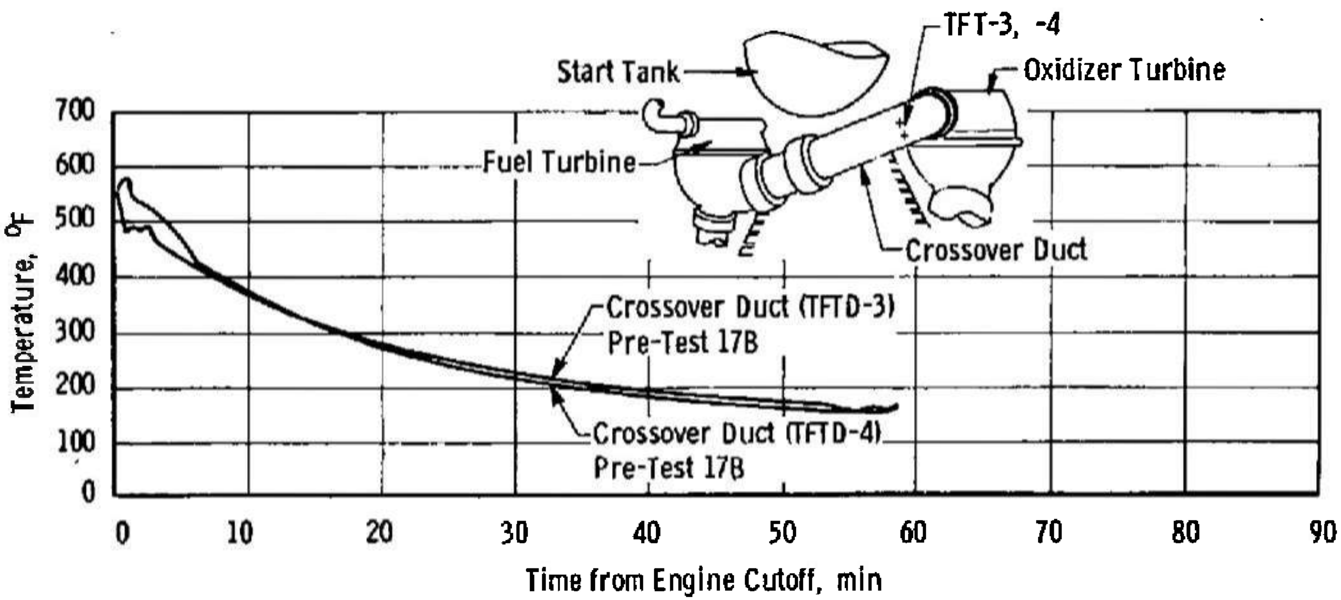


b. Gas Generator Transient Pressures

Fig. 30 Start Transient Pressures, Test 15B

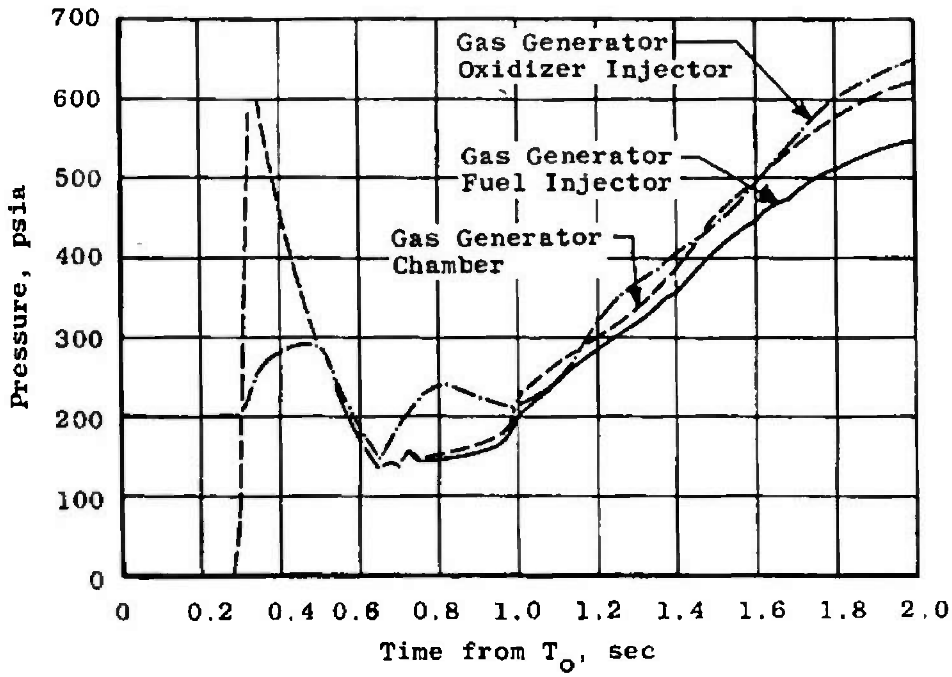


a. Turbine Inlet Temperatures

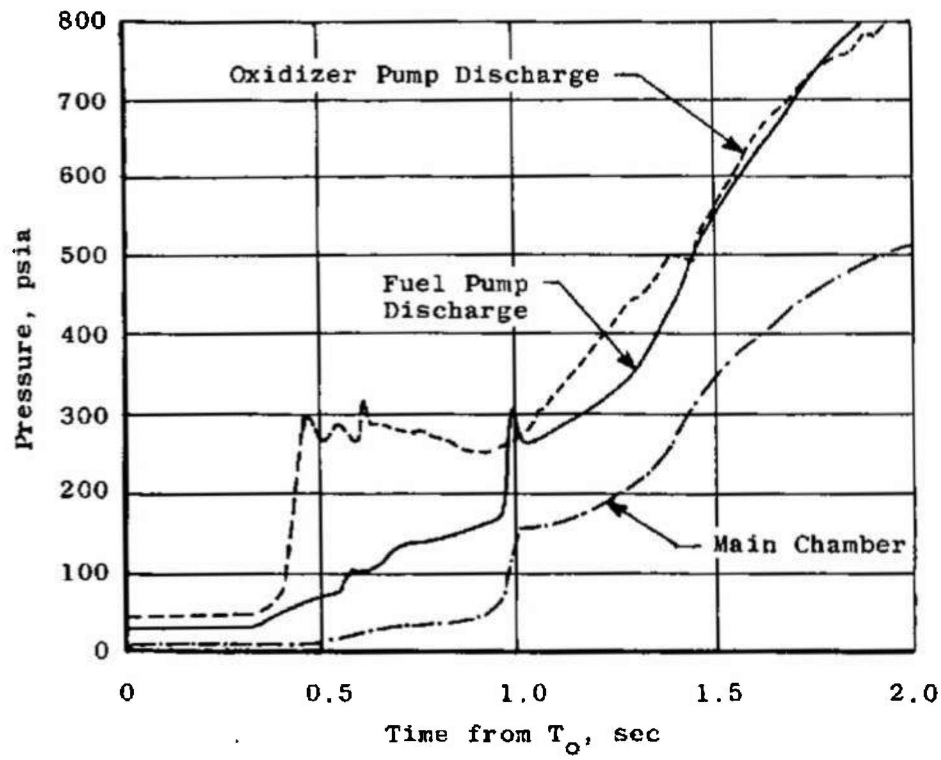


b. Crossover Duct Skin Temperatures

Fig. 31 Turbine Hardware Temperatures, AS 203 and Test 17B



a. Gas Generator Transient Pressures



b. Propellant System Transient Pressures

Fig. 32 Start Transient Pressures, Test 17B

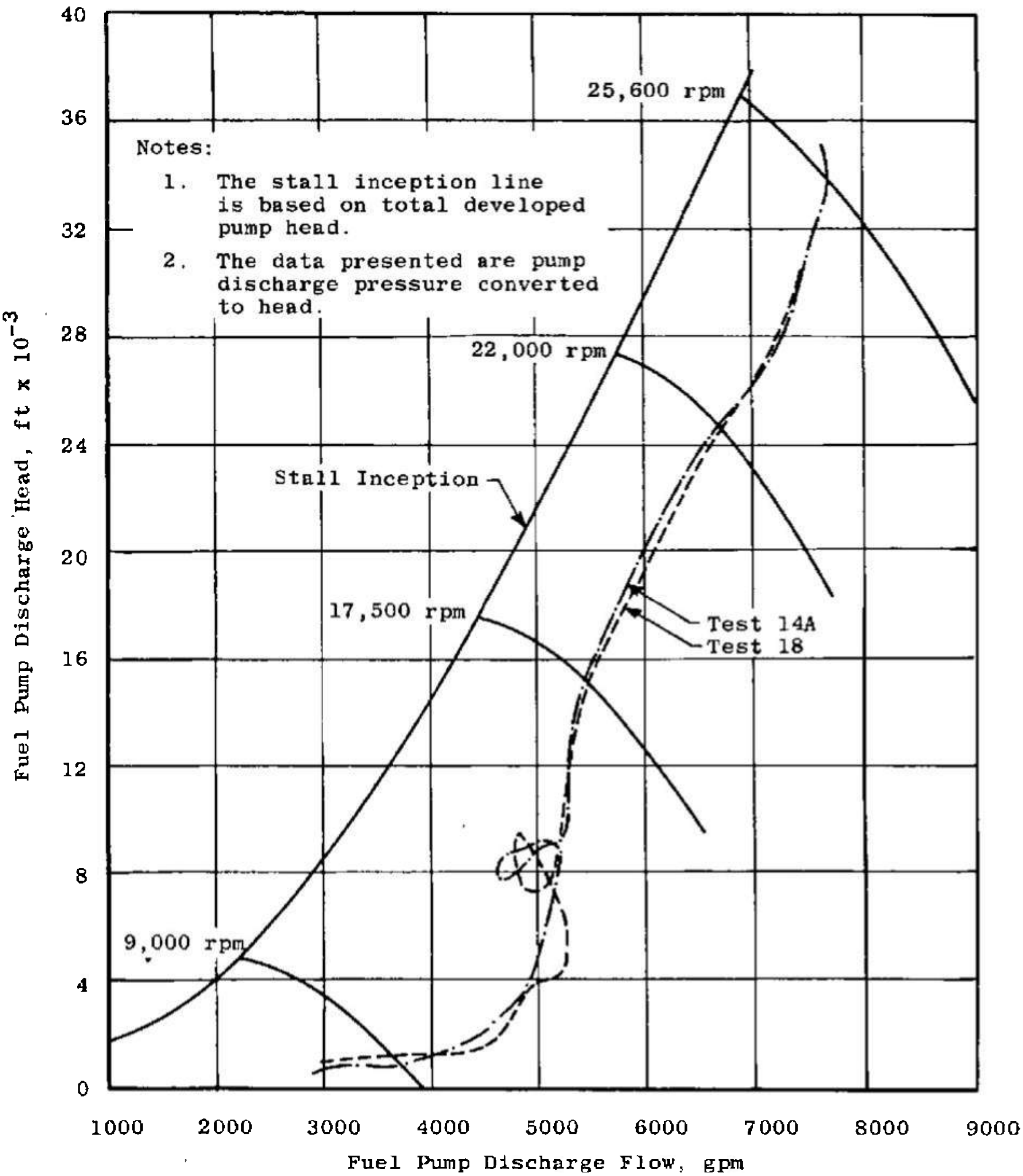


Fig. 33 Fuel Pump Start Transient Performance

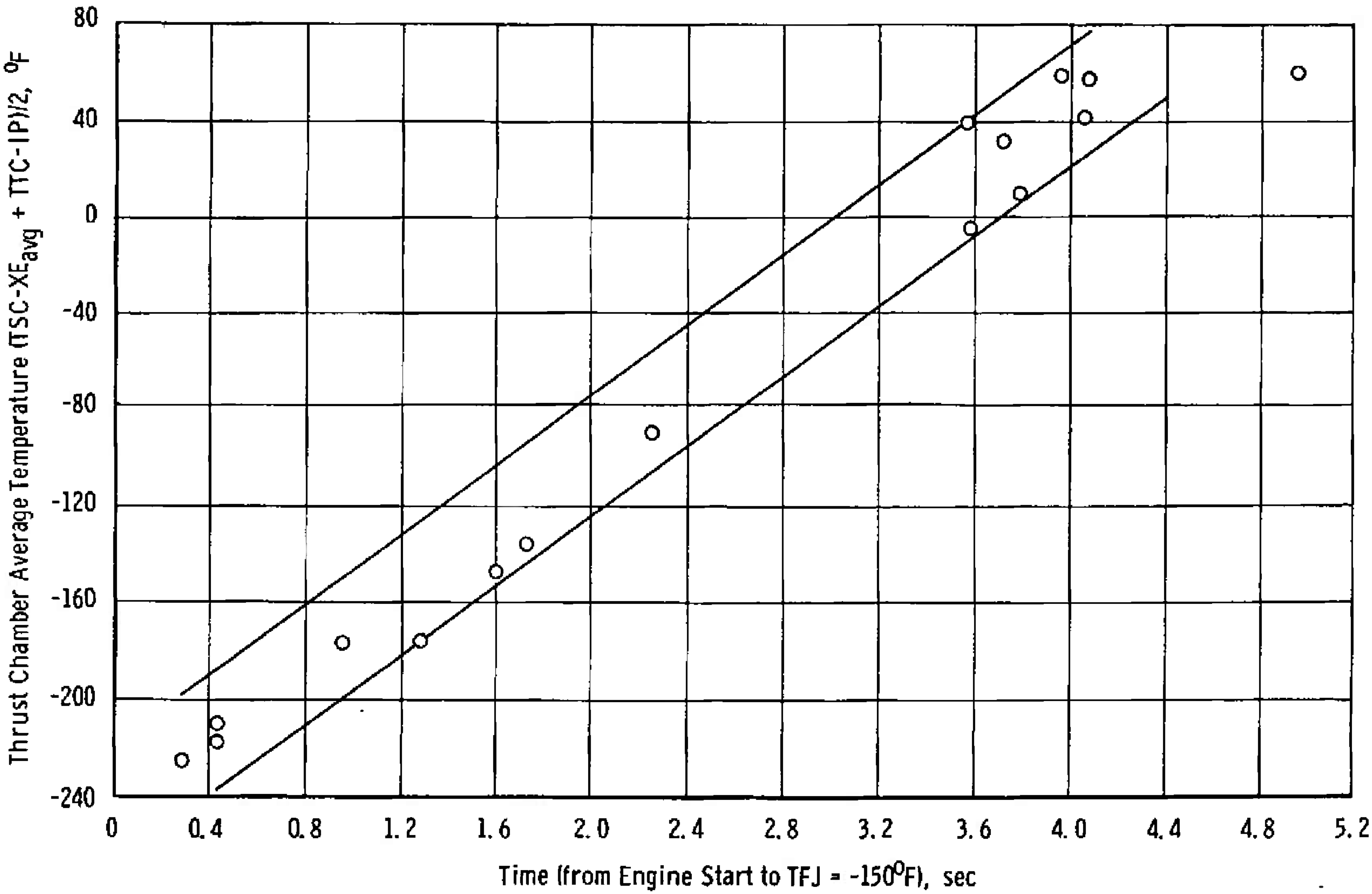


Fig. 34 Time from ES to Fuel Injector Temperature of -150°F

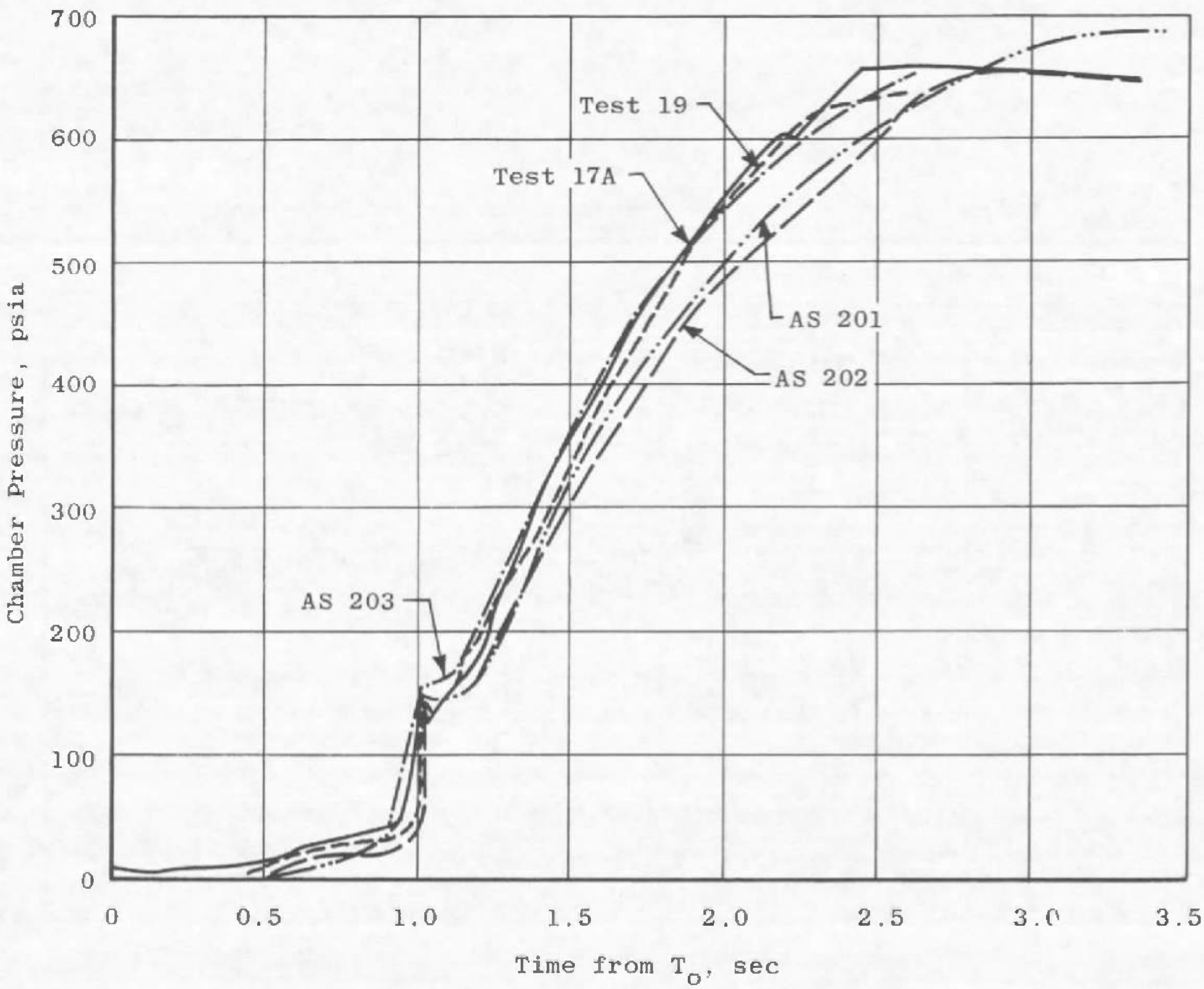


Fig. 35 Chamber Pressure, AEDC and Flight

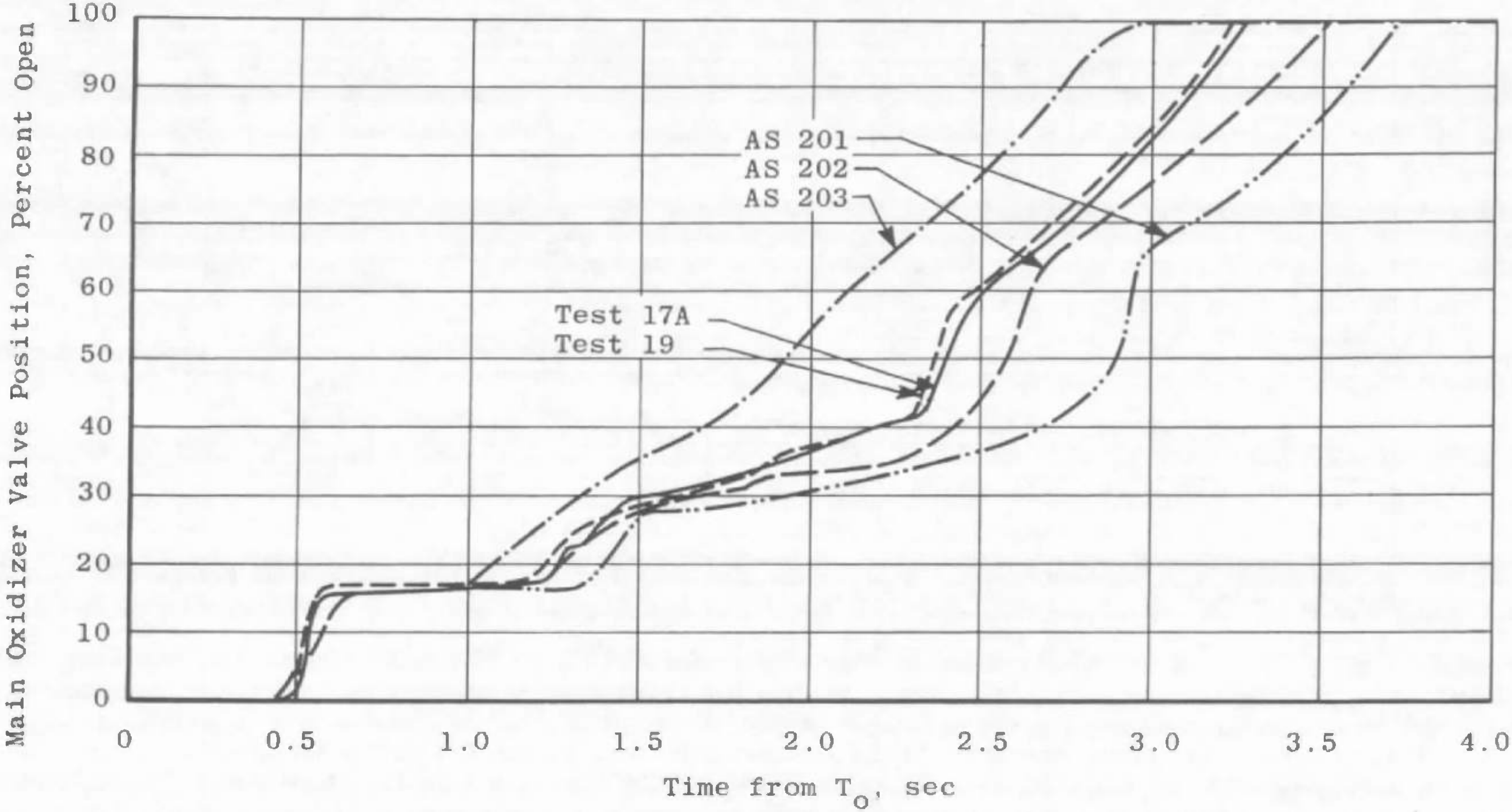


Fig. 36 Position of MOV, AEDC and Flight

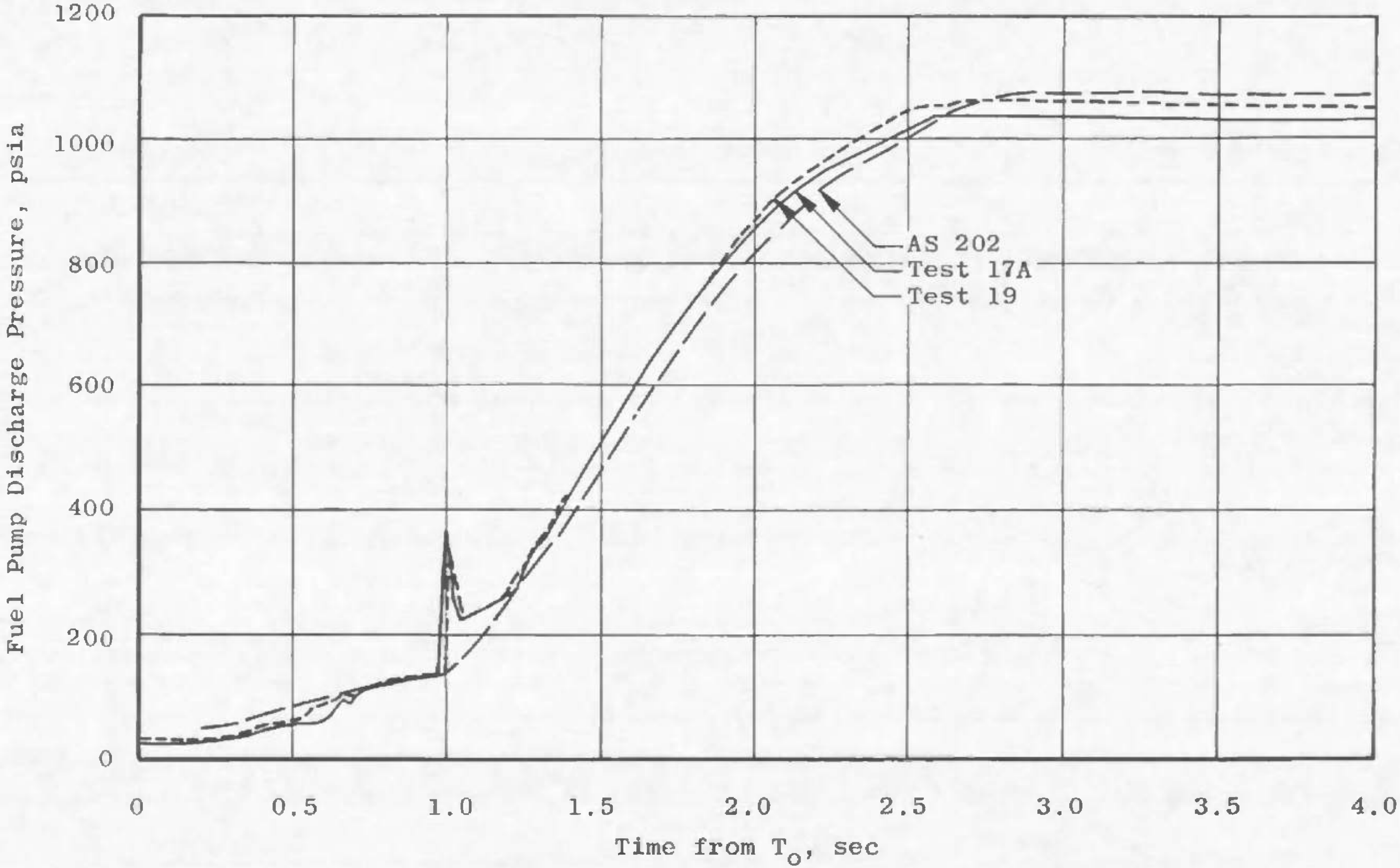


Fig. 37 Fuel Pump Discharge Pressures, AEDC and Flight

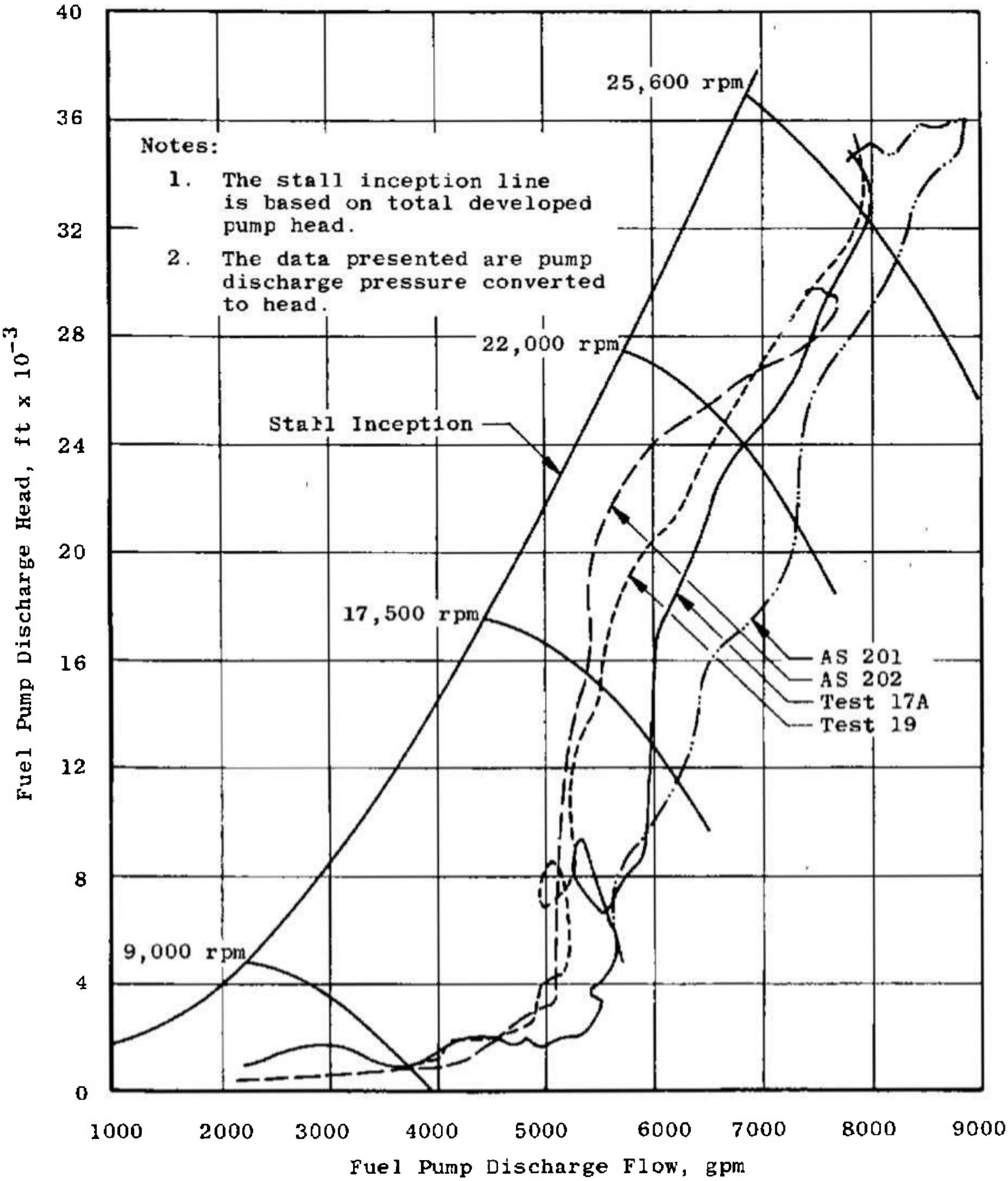


Fig. 38 Fuel Pump Start Transient Performance, AEDC and Flight

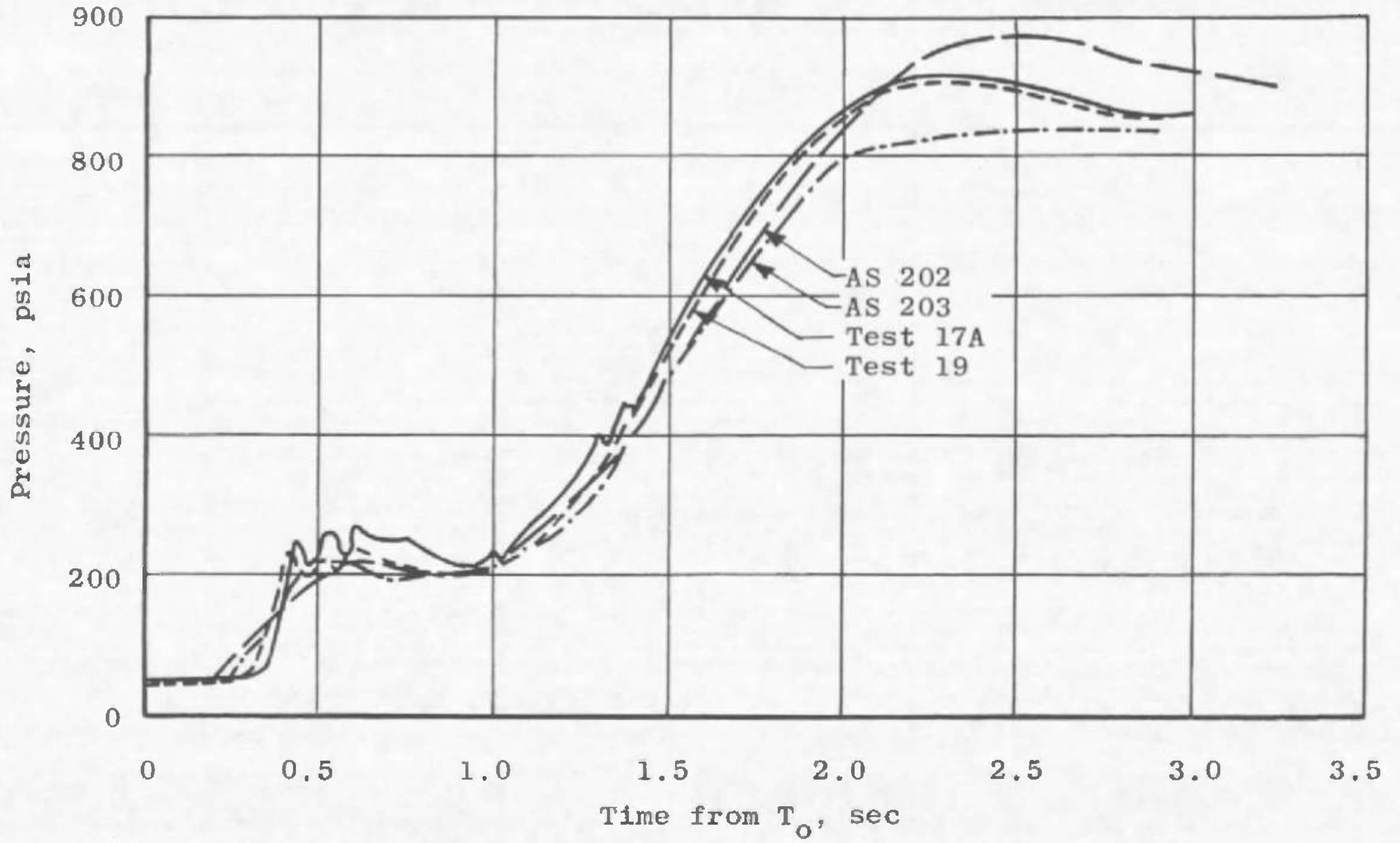
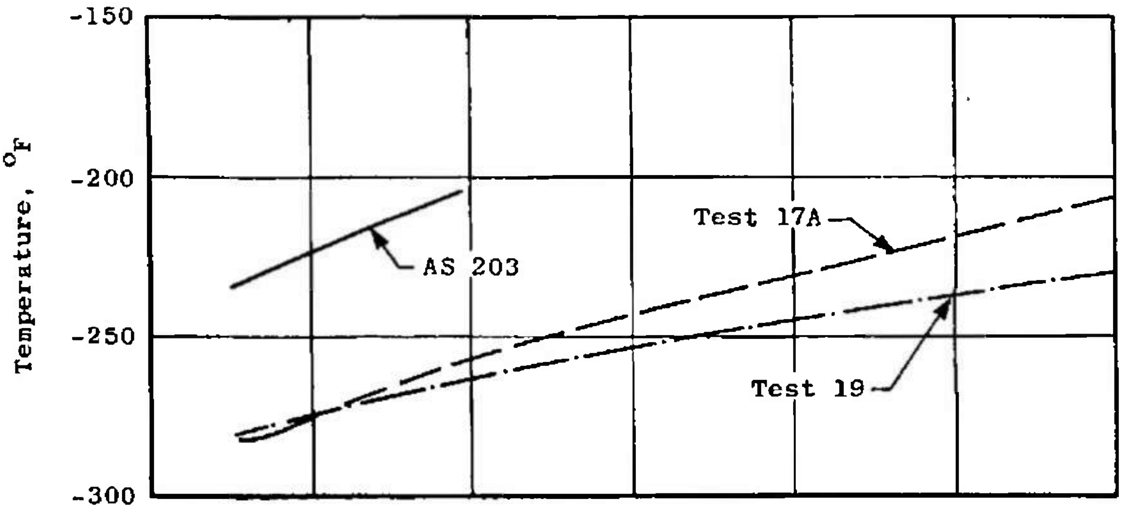
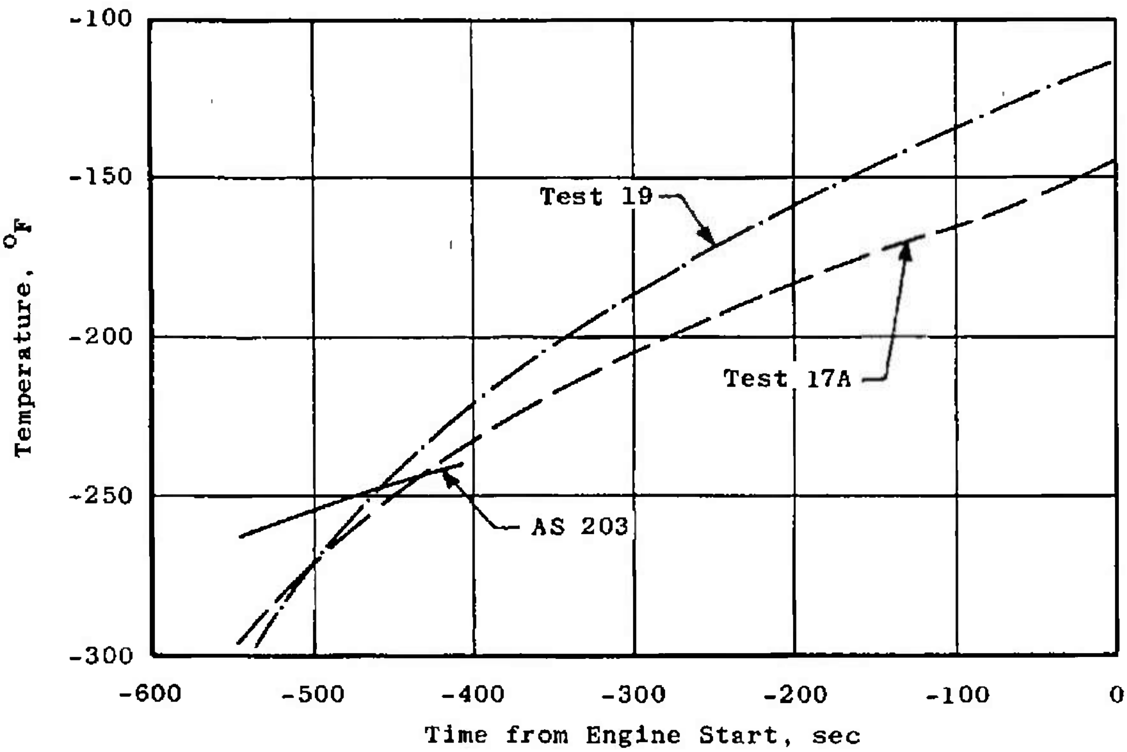


Fig. 39 Oxidizer Pump Discharge Pressures, AEDC and Flight



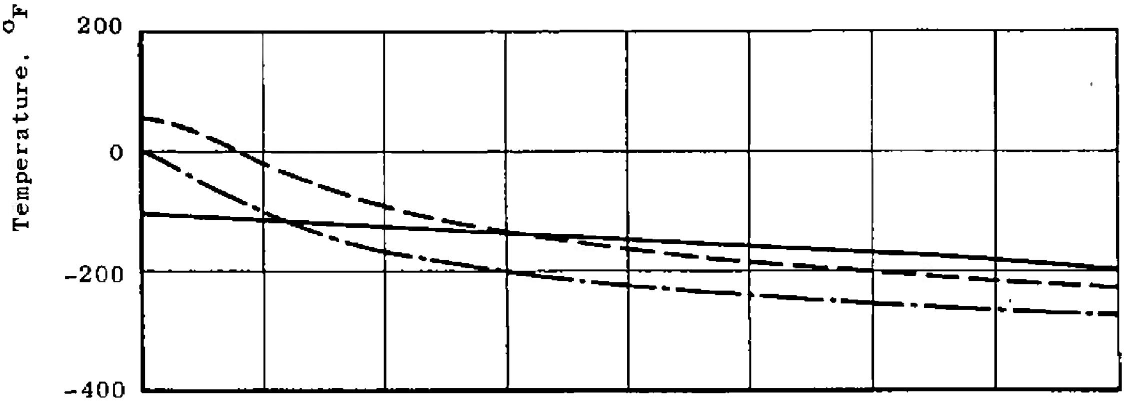
a. Thrust Chamber Throat Temperatures



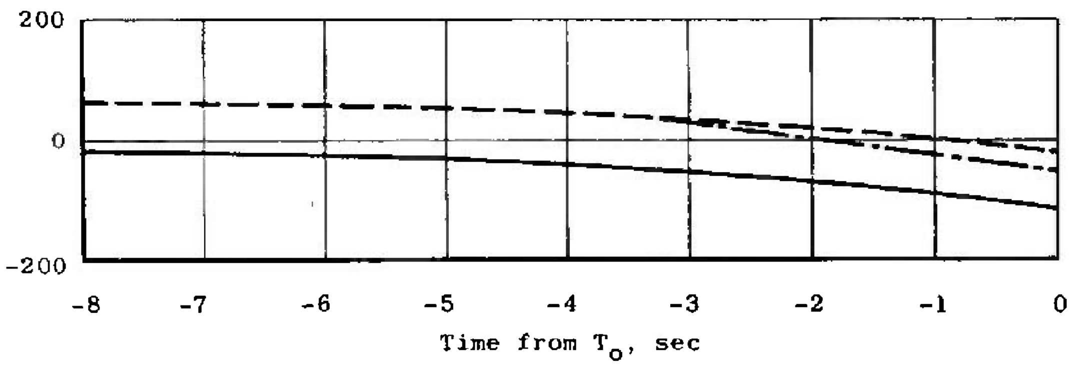
b. Thrust Chamber Exit Temperatures

Fig. 40 Thrust Chamber Temperatures during Boost Phase Warmup, AEDC and Flight

a. Fuel Injector Temperatures

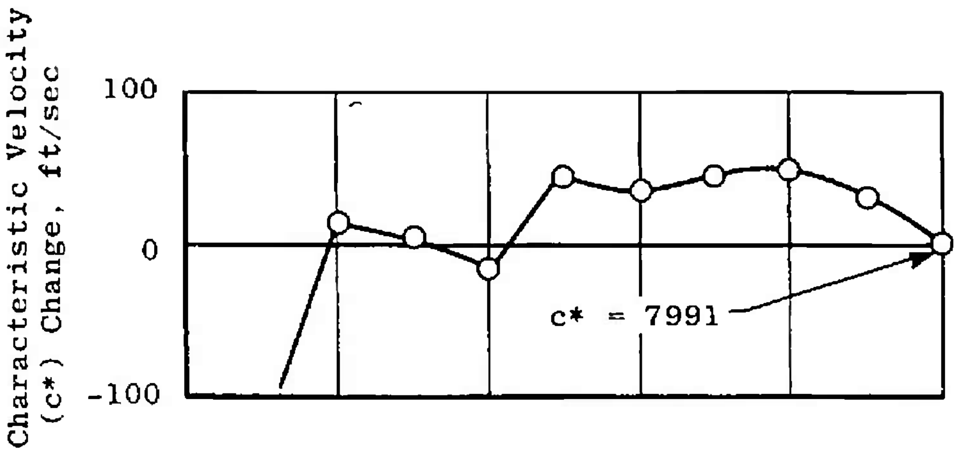


b. Thrust Chamber Exit Temperatures

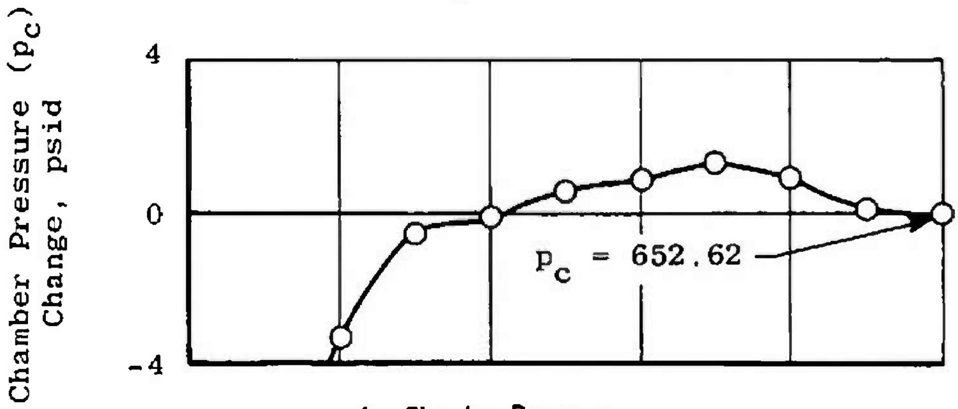


c. Thrust Chamber Throat Temperatures

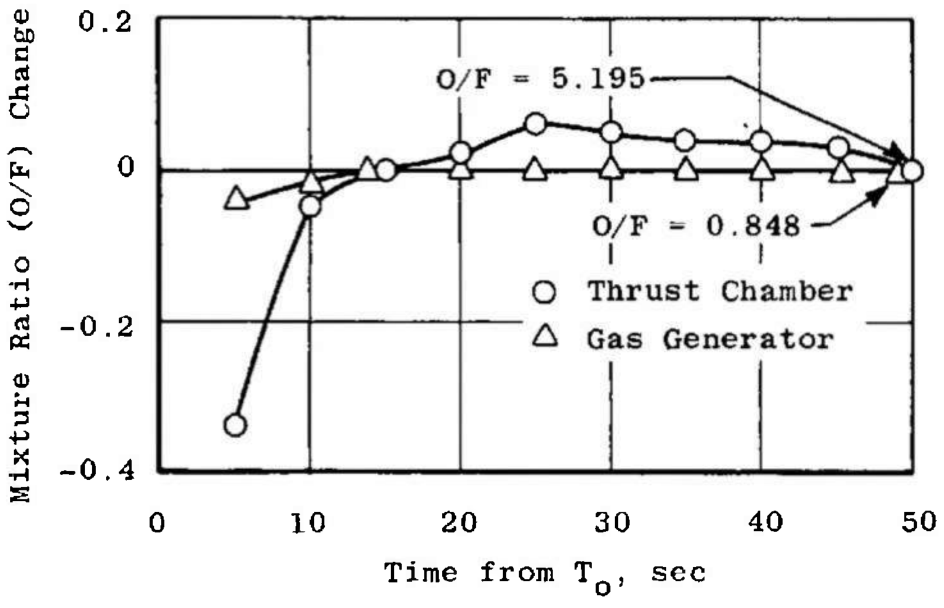
Fig. 41 Thrust Chamber Temperatures during 8-sec Fuel Lead, AEDC and Flight



a. c*

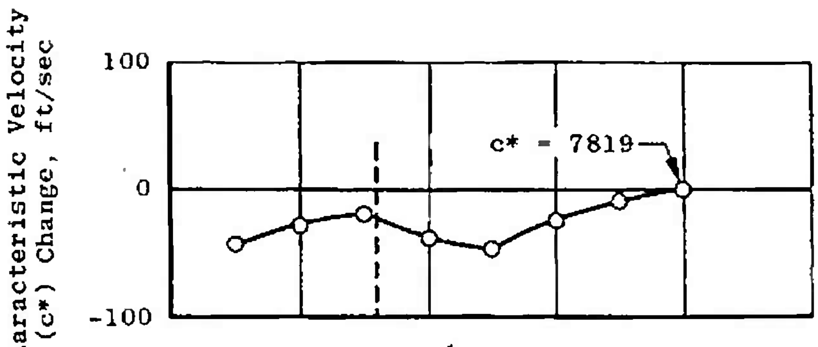


b. Chamber Pressure

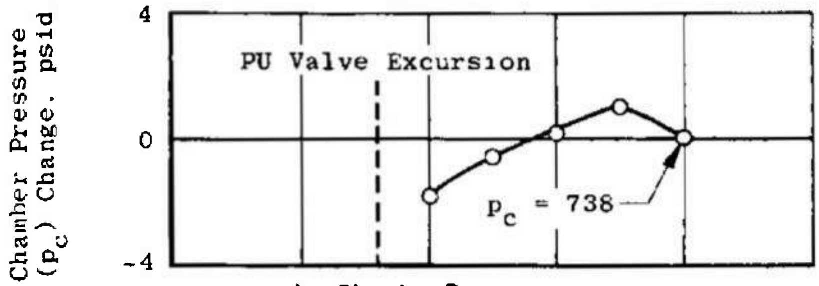


c. Mixture Ratio

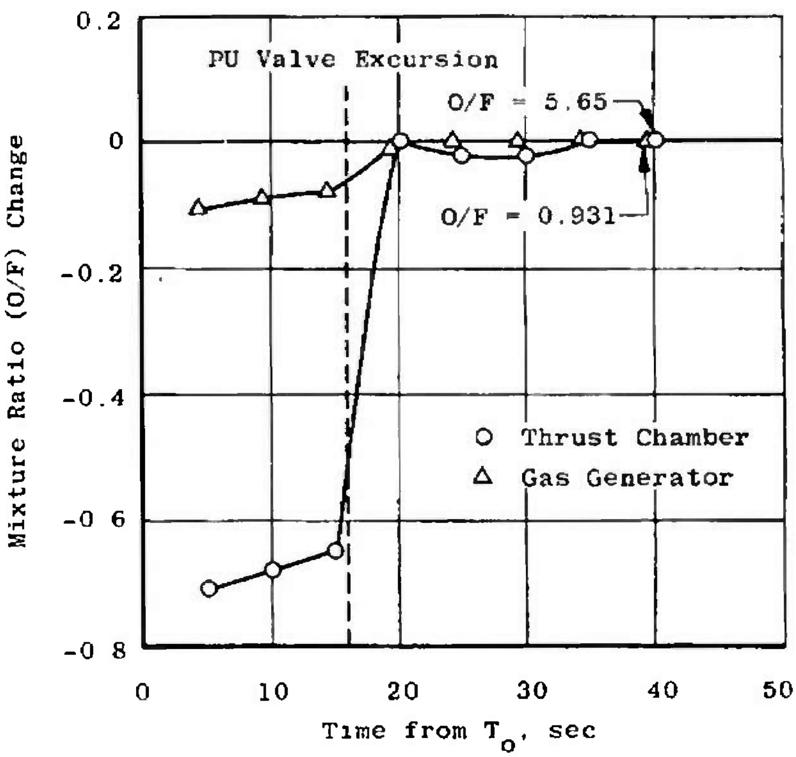
Fig. 42 Performance Variation during Main Stage, Test 12B



a. c*

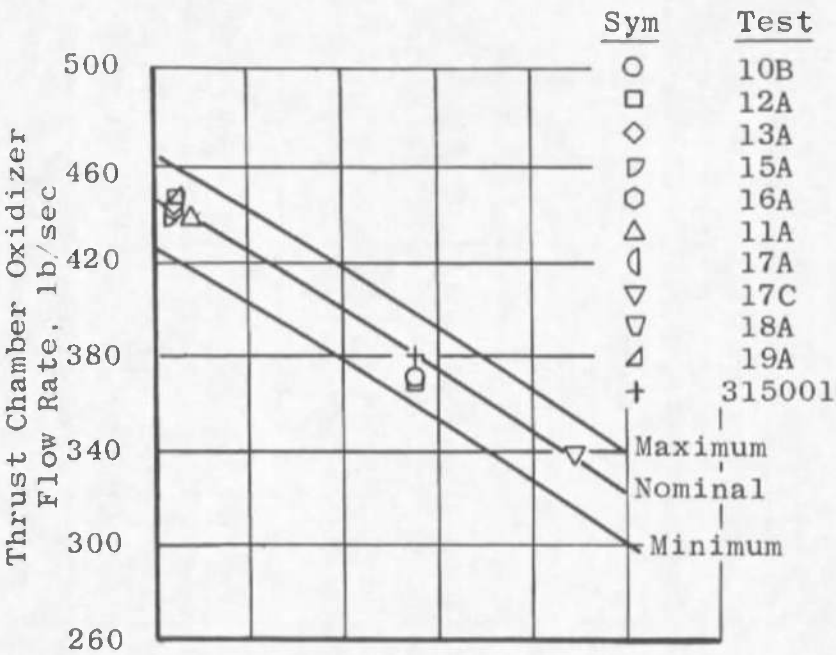


b. Chamber Pressure

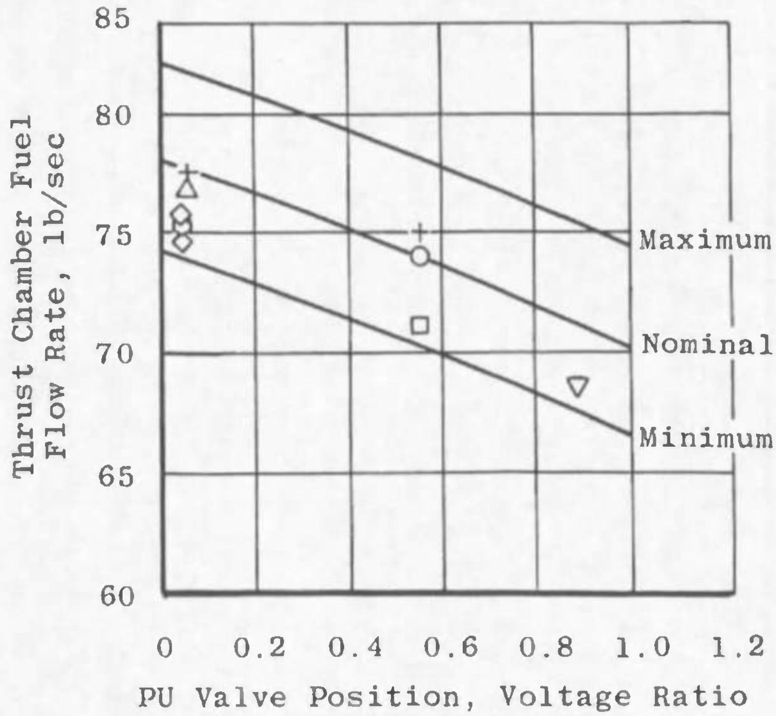


c. Mixture Ratio

Fig. 43 Performance Variation during Main Stage, Test 15A

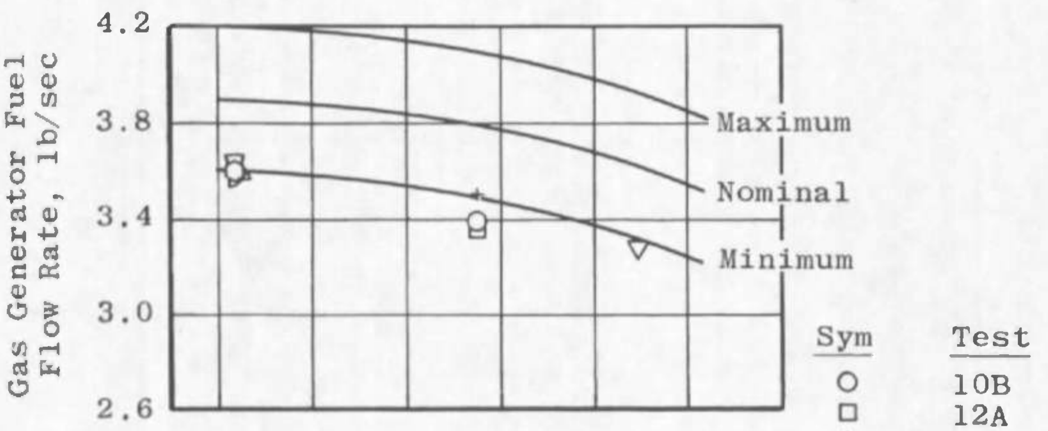


a. Oxidizer Flow Rate

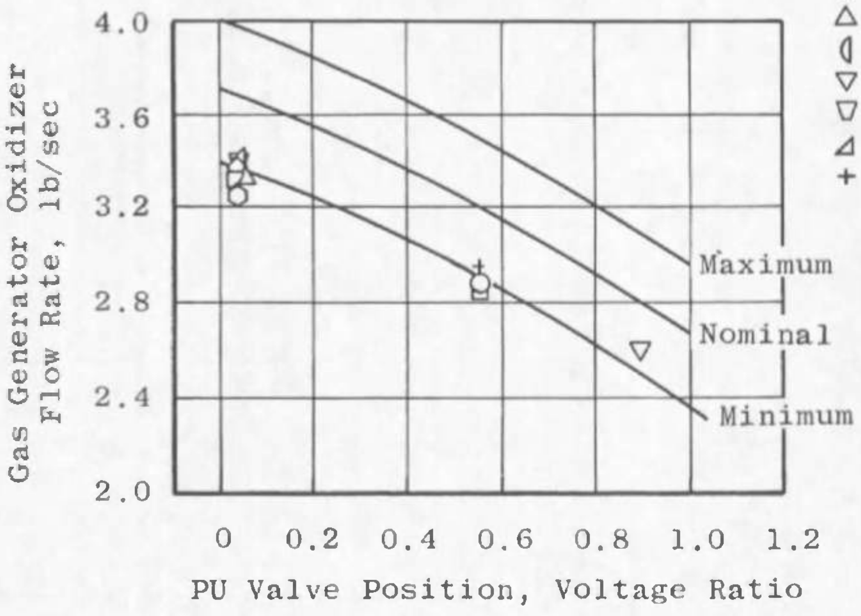


b. Fuel Flow Rate

Fig. 44 Thrust Chamber Flow Rates

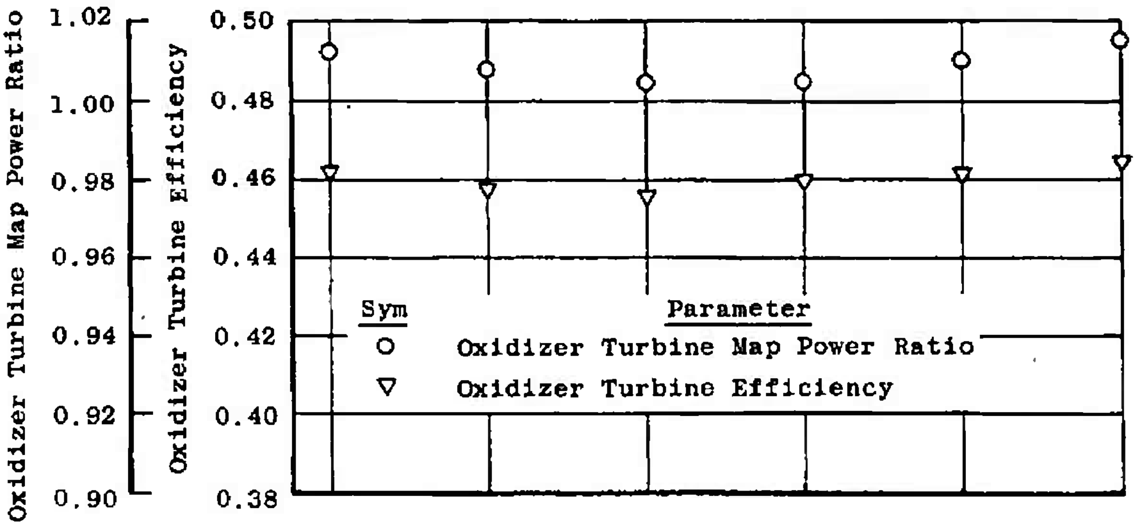


a. Fuel Flow Rate

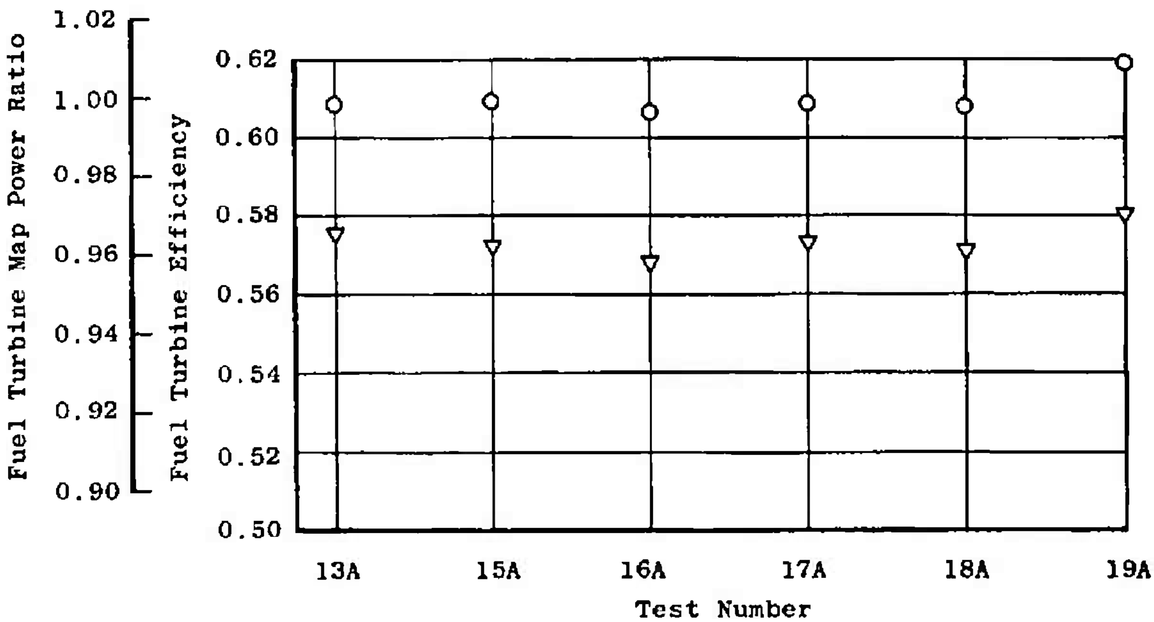


b. Oxidizer Flow Rate

Fig. 45 Gas Generator Flow Rates

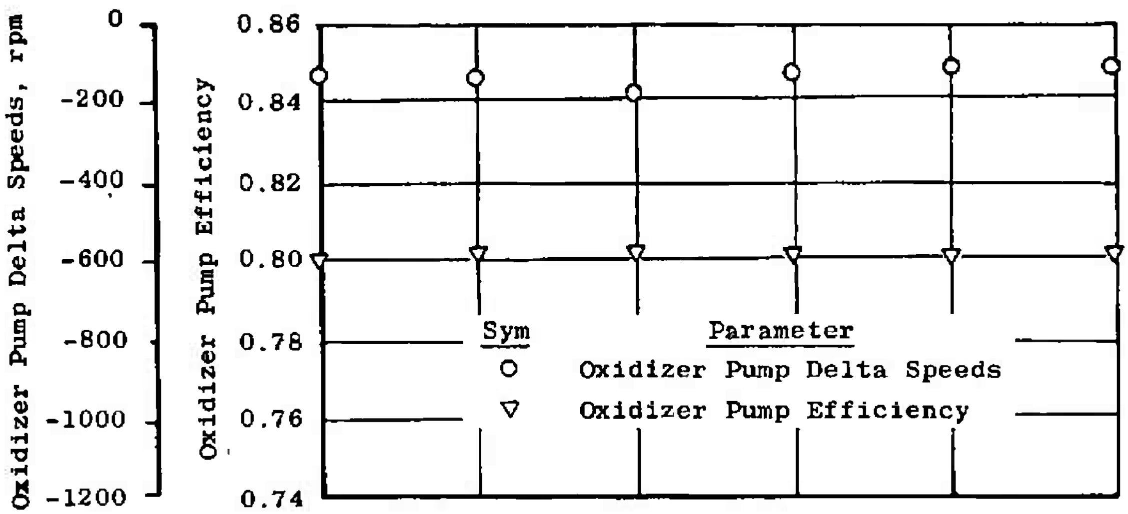


a. Oxidizer Turbine

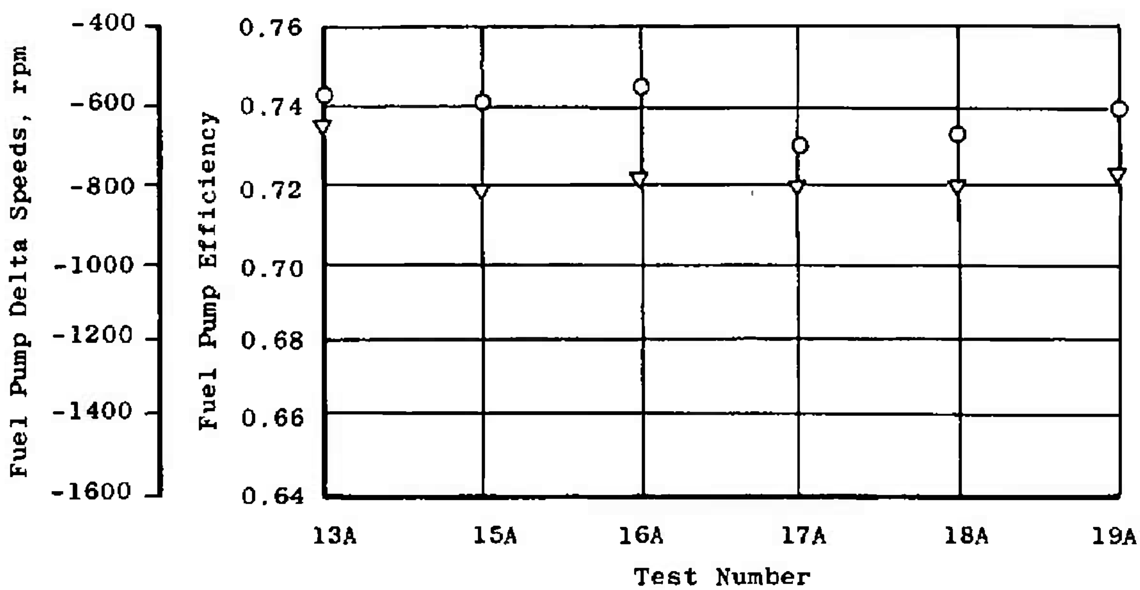


b. Fuel Turbine

Fig. 46 Oxidizer and Fuel Turbine Performance (Time Interval, 29.5 ± 0.5 sec)



a. Oxidizer Pump



b. Fuel Pump

Fig. 47 Oxidizer and Fuel Pump Performance (Time Interval, 29.5 ± 0.5 sec)

TABLE I
NOMINAL PERFORMANCE OF THE J-2 ENGINE

Engine	
Thrust, Vacuum, lb	225,000
Mixture Ratio, O/F	5.50
Total Propellant Flow Rate, lb/sec	531.8
Chamber Pressure, psia	778
Oxidizer Turbopump	
Pump	
Inlet Pressure, psia	39.00
Discharge Pressure, psia	1081
Flow Rate, lb/sec	458.6
Power, bhp	2202
Speed, rpm	8572
Turbine	
Inlet Pressure (Total), psia	85.9
Outlet Pressure (Static), psia	32.2
Inlet Temperature, °F	770
Outlet Temperature, °F	612
Fuel Turbopump	
Pump	
Inlet Pressure, psia	30.00
Discharge Pressure, psia	1225
Flow Rate, lb/sec	82.5
Power, bhp	7739
Speed, rpm	26702
Turbine	
Inlet Pressure (Total), psia	633.6
Outlet Pressure (Static), psia	87.1
Inlet Temperature, °F	1200
Outlet Temperature, °F	770
Gas Generator	
Chamber Pressure, psia	654.7
Mixture Ratio, O/F	0.939
Total Propellant Flow Rate, lb/sec	7.02
Tank Pressurization Flow Rates	
Hydrogen, lb/sec	0.80
Oxygen, lb/sec	1.8

TABLE II
MAJOR J-2 ENGINE S/N J-2052 COMPONENTS
(EFFECTIVE TEST J4-1554-12)

Part Name	P/N	S/N
Thrust Chamber Body	206600-31	4076553
Thrust Chamber Injector Assembly	208021-11	4071189
Fuel Turbopump Assembly	459000-121	4078258
Oxidizer Turbopump Assembly	458175-71	6623549
Start Tank	303439	0064
ASI	206280-21	4080118
GG Oxidizer Injector and Poppet Assembly	303323	4076827
GG Fuel Injector and Combustor	308360-11	2008734
Pneumatic Control Assembly	556947	4078075
Electrical Control Package	502670	4078604
Primary Flight Instrumentation Package	703685	4078716
Auxiliary Flight Instrumentation Package	703680	4078718
Main Fuel Valve	409920	4078041
MOV	410431	4072882
GG Control Valve	309040	4078292
STDV	306875	4062591
Oxidizer Turbine Bypass Valve	409930	4067221
PU Valve	251351-11	4068944
Main-Stage Control Valve	558069	8320433
Ignition Stage Control Valve	558069	8320432
He Control Valve	NA5-27273	340921
Start Tank Vent and Relief Valve	557838	4074529
He Tank Vent Valve	NA5-27273	340916
Fuel Bleed Valve	309034	4077749
Oxidizer Bleed Valve	309029	4077746
ASI Oxidizer Valve	308880	4077205
P/A Purge Control Valve	557823	4068834
Start Tank Fill/Refill Valve	558000	4079001
Fuel Flowmeter	251225	4077752
Oxidizer Flowmeter	251216	4074114
Fuel Injector Temperature Transducer	NA5-27441	12775
Restartable Ignition Detect Probe	NA5-27298T2	107

TABLE III
CONFIGURATION CHANGE RECORD FOR J-2 ENGINE S/N J-2052
(EFFECTIVE TESTS J4-1554-12 THROUGH J4-1554-19)

Date	Configuration Change
11/17/66	Addition of Thermocouple per ECP* J2-564/Mod 253
11/21/66	Accelerometer (P/N ES6412B, S/N 407) Installed
11/22/66	Installation of Thrust Chamber Insulation per ECP* J2-474R5/Mod 245
11/28/66	Addition of Critical Component Thermocouples per RFD** 6-66
11/28/66	Black Painting of Thrust Chamber Insulation per RFD** 7-66
12/1/66	Accelerometer (P/N ES6412A, S/N 384) Installed
12/2/66	Test J4-1554-12
12/7/66	Accelerometer (P/N ES6412, S/N 564) Installed
12/7/66	Accelerometer (P/N ES6412, S/N 275) Installed
12/10/66	GG Control Valve (P/N 309040 S/N 4055754) Installed
12/10/66	GG Control Valve Position Indicator (P/N 2001611001, S/N 7-1836) Installed
12/13/66	Test J4-1554-13
12/14/66	GGOT Transducer (P/N NA5-27342T4-1, S/N 511) Installed
12/20/66	Test J4-1554-14
12/30/66	Installation of MOV Closing Control Line Thermocouple per RFD** 8-66

*Rocketdyne Engineering Change Proposal

**Rocketdyne Field Directive

TABLE III (Continued)

Date	Configuration Change
1/4/67	Test J4-1554-15
1/6/67	Incorporation of Double-Weld Shotpeened Flange to Tube Joint-Main Fuel and Oxidizer Turbine Bypass Valves Control Line per ECP* J2-544/Mod 236
1/6/67	Deletion of ASI Chamber Pressure Static Stage Instrumentation Line per ECP* J2-437-1/Mod 246
1/13/67	Test J4-1554-16
1/13/67	Spark Assembly Igniter (P/N 206280-21, S/N 4084017) Installed
1/13/67	Fuel Turbine Inlet Temperature Transducer (P/N NA5-27323T3, S/N 272) Installed
1/14/67	Addition of Accelerometer Mount on Oxidizer Dome per ECP* J2-549/Mod 247
1/14/67	Installation of Turbine Crossover Duct Flight Temperature Transducer per ECP* J2-547/Mod 241
1/16/67	Start Tank Vent and Relief Valve (P/N 557828-X2, S/N 4046446) Installed
1/16/67	Installation of Air Filler Valves on ASI and GG Spark Igniter Cable Assembly per ECP* J2-538/Mod 244
1/19/67	Test J4-1554-17
1/20/67	Restartable Ignition Detect Probe (P/N NA5-27298T2, S/N 213) Installed
1/20/67	GGOT Transducer (P/N NA5-27342T4-1, S/N 506) Installed
1/23/67	Addition of Spare Turbine Crossover Duct Thermocouples per RFD** 5-67
1/23/67	Installation of MOV Closing Control Line Pressure Monitor per RFD**6-67
1/25/67	Oxidizer Turbine Bypass Valve (P/N 409930, S/N 4081832) Installed

*Rocketdyne Engineering Change Proposal
**Rocketdyne Field Directive

TABLE III (Concluded)

Date	Configuration Change
1/25/67	Oxidizer Turbine Bypass Valve Position Indicator (P/N 408012A, S/N 8288303) Installed
1/26/67	Test J4-1554-18
1/27/67	Addition of Thrust Chamber Thermocouples per RFD** 7-67
1/27/67	Installation of Thrust Chamber Heater Blanket per RFD** 8-67
1/27/67	Addition of GG Valve Balance Line Purge per RFD** 9-67
1/30/67	Installation of He Regulator Chilling System per RFD** 10-67
1/31/67	Inspection of Start Tank Weld per RFD** 11-67
2/2/67	He Control Solenoid (P/N NA5-27273, S/N 335164) Installed
2/5/67	Test J4-1554-19

**Rocketdyne Field Directive

TABLE IV
INSTRUMENTATION SUMMARY

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Temperature			°F						
1	Fuel Pump Inlet	TFPI-1		-425 to -415	x					
2	Fuel Pump Inlet	TFPI-2		-425 to -400	x	x		x		
3	Oxidizer Pump Inlet	TOPI-1		-300 to -275	x					
4	Oxidizer Pump Inlet	TOPI-2		-310 to -270	x	x		x		
5	Fuel Pump Discharge	TFPD-1P	PFT1	-425 to -400	x	x	x			
6	Fuel Pump Discharge	TFPD-2	PFT1	-425 to -400	x					
7	Oxidizer Pump Discharge	TOPD-1P	POT3	-300 to -250	x		x	x		
8	Oxidizer Pump Discharge	TOPD-2	POT3	-300 to -250	x					
9	Fuel Bleed Valve	TFBV-1A		-425 to -375	x					
10	Fuel Bleed Valve	TFBV-2		-425 to -375	x					
11	Oxidizer Bleed Valve	TOBV-1A		-300 to -250	x					
12	Oxidizer Bleed Valve	TOBV-2		-300 to -250	x					
13	Main Fuel Injector	TFJ-1P	CFT2	-425 to +100	x	x	x			
14	ASI Fuel Injector	TFAS1J	IFT1	-425 to +100	x		x			
15	ASI Oxidizer Injector	TOAS1J	IOT1	-300 to +100	x		x			
16	Fuel Turbine Inlet	TFTI-1P	TGT1	0 to 1800	x			x		
17	GG Overtemperature	TGGO-1A	GGT1	0 to 2000	x		x			
18	Fuel Turbine Outlet	TFTO	TGT2	0 to 1800	x					Added on Test 13
19	Oxidizer Turbine Inlet	TOTI-1P	TGT3	0 to 1200	x		x			
20	Oxidizer Turbine Outlet	TOTO-1P	TGT4	0 to 1000	x					
21	Fuel Pump Bearing	TFPB-1A		-425 to -325	x					
22	Oxidizer Pump Bearing	TOPB-1A		-300 to -250	x					
23	Oxidizer Valve Act. Cap	TSOVC		-325 to 150	x					
24	Fuel Start Tank	TFST-1P	TFT1	-350 to +100	x					

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Temperature			°F						
25	Fuel Start Tank	TFST-2	TFT1	-350 to +100						
26	He Tank	THET-1P	NNT1	-350 to +100	x					
27	Electrical Control Package	TECP-1P	NST1A	-300 to +200	x			x		
28	Electrical Control Package	TECP-2	NST1B	-300 to +200	x					
29	Primary Instrument Package	TPIP-1P		-300 to +200	x					
30	Auxiliary Instrument Package	TAIP-1P		-300 to +200	x					
31	Thrust Chamber Jacket	TTC-1P	CS1	-425 to +100	x			x		
32	Thrust Chamber Jacket	TTC-2	CS1A	-425 to +100	x					
33	Thrust Chamber Skin	TSC-1T		-300 to +300	x					
34	Thrust Chamber Skin	TSC-1E		-300 to +300	x					
35	Thrust Chamber Skin	TSC-2E		-300 to +300	x					
36	Thrust Chamber Skin	TSC-3T		-300 to +300	x					
37	Thrust Chamber Skin	TSC-3E		-300 to +300	x					
38	Thrust Chamber Skin	TSC-4T		-300 to +300	x					
39	Thrust Chamber Skin	TSC-4E		-300 to +300	x					
40	Thrust Chamber Skin	TSC-5T		-300 to +300	x					
41	Thrust Chamber Skin	TSC-5E		-300 to +300	x					
42	Thrust Chamber Skin	TSC-6T		-300 to +300	x					
43	Thrust Chamber Skin	TSC-6E		-300 to +300	x					
44	Thrust Chamber Skin	TSC-7E		-300 to +300	x			x		
45	Thrust Chamber Skin	TSC-8T		-300 to +300	x					
46	Thrust Chamber Skin	TSC-8E		-300 to +300	x					
47	Oxidizer Recirculation Pump Return Line	TORPR		-300 to -140	x					

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Temperature			°F						
48	Fuel Recirculation Pump Return Line	TFRPR		-425 to -250	x					
49	Oxidizer, Recirculation Pump Outlet	TORPO		-300 to -250	x					
50	Fuel Recirculation Pump Outlet	TFRPO		-425 to -410	x					
51	Fuel Run Tank	TFRT-1		-425 to -413	x					
52	Fuel Run Tank	TFRT-2		-425 to -413	x					
53	Oxidizer Run Tank	TORT-1		-300 to -287	x			x		
54	Oxidizer Run Tank	TORT-3		-300 to -287	x					
55	Fuel Tank Pressure Vent Line	TFVL		-400 to +100	x					
56	He Regulator Body	TBHR-1		-200 to +400	x			x		Added on Test 19
57	He Regulator Body	TBHR-2		-200 to +400	x			x		Added on Test 19
58	Fuel Pump Discharge Duct	TFPDD		-320 to 300	x					
59	Fuel Turbine Discharge Collector	TFTD-1		-200 to 800	x					
60	Fuel Turbine Discharge Collector	TFTD-2		-200 to 800	x					
61	Fuel Turbine Discharge Line	TFTD-3		-200 to 1000	x			x		
62	Fuel Turbine Discharge Line	TFTD-4		-200 to 1000	x			x		
63	Thrust Chamber Exit	TTCEP-1		-425 to 100	x					
64	Thrust Chamber Exit	TTCEP-2		-425 to 100	x					
65	Oxidizer Bootstrap Line	TOBS		-300 to 500	x					
66	Thrust Chamber Skin	TSC9-1		-300 to 500	x					
67	Thrust Chamber Skin	TSC9-2		-300 to 500	x					
68	Thrust Chamber Skin	TSC9-3		-300 to 500	x					
69	Thrust Chamber Skin	TSC10-1		-300 to 500	x					
70	Thrust Chamber Skin	TSC10-2		-300 to 500	x					
71	Thrust Chamber Skin	TSC10-3		-300 to 500	x					

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Temperature			°F						
72	Thrust Chamber Skin	TSC11-1		-300 to 500	x					
73	Thrust Chamber Skin	TSC11-2		-300 to 500	x					
74	Thrust Chamber Skin	TSC11-3		-300 to 500	x					
75	Thrust Chamber Skin	TSC12-1		-300 to 500	x					
76	Thrust Chamber Skin	TSC12-2		-300 to 500	x					
77	Thrust Chamber Skin	TSC12-3		-300 to 500	x					
78	Thrust Chamber Skin	TSC13-1		-300 to 500	x					
79	Thrust Chamber Skin	TSC13-2		-300 to 500	x					
80	Thrust Chamber Skin	TSC13-3		-300 to 500	x					
81	Thrust Chamber Skin	TSC14-1		-300 to 500	x					
82	Thrust Chamber Skin	TSC14-2		-300 to 500	x					
83	Thrust Chamber Skin	TSC14-3		-300 to 500	x					
84	STDV	TVBST-D		-300 to 500	x					
85	Start Tank Fill Valve	TVBST-F		-300 to 500	x					
86	STDV Solenoid	TVBST-S		-300 to 500	x					
87	Start Tank Vent Valve	TVBST-V		-300 to 500	x					
88	LO_2 Valve Actuator Line Skin Temperature	TSOVAL-2		-325 to 150	x					Added on Test 18
89	Cell	TA-1		-50 to 800	x					
90	Cell	TA-2		-50 to 800	x					
91	Cell	TA-3		-50 to 800	x					
92	Cell	TA-4		-50 to 800	x					
93	Thrust Chamber Skin	TSC-15		-300 to 500	x					Added on Test 19
94	Thrust Chamber Skin	TSC-16		-300 to 500	x					Added on Test 19

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Temperature			°F						
95	Thrust Chamber Skin	TSC-17		-300 to 500	x					Added on Test 19
96	Thrust Chamber Skin	TSC-18		-300 to 500	x					Added on Test 19
97	LO_2 Valve Actuator Line Skin	TSOVAL-1		-200 to 100	x			x		Added on Test 15
98	Fuel Turbine Discharge Collector	TFTD-1R		-360 to 1200	x					Added on Test 17
99	Fuel Turbine Discharge Collector	TFTD-2R		-360 to 1200	x					Added on Test 17
100	Fuel Turbine Discharge Collector	TFTD-3R		-360 to 1200	x			x		Added on Test 17
101	Fuel Turbine Discharge Collector	TFTD-4R		-360 to 1200	x					Added on Test 17
	Pressure			psia						
102	Thrust Chamber Pressure	PC-1P	CG1	0 to 1000	x					
103	Thrust Chamber Pressure	PC-2	CG1	0 to 1000	x	x				
104	Thrust Chamber Pressure	PC-3	CG1A	0 to 1000	x		x			
105	ASI Chamber	PCAS1	IG1	0 to 1000	x		x			Deleted after Test 17
106	Fuel Jacket Inlet Manifold	PFMI	CF1	0 to 2000	x					
107	GG Oxidizer Bleed Valve	POBV		0 to 1500	x					
108	GG Fuel Injector	PFJGG-1A	GF4	0 to 1000	x	x				
109	GG Fuel Injector	PFJGG-2	GF4	0 to 1000	x		x			
110	GG Oxidizer Injector	POJGG-1A	GO5	0 to 1000	x	x	x			
111	GG Oxidizer Injector	POJGG-2	GO5	0 to 1000	x					
112	GG Chamber	PCGG-1P	GG1	0 to 1000	x	x	x			
113	GG Chamber	PCGG-2	GG1A	0 to 1000	x					
114	Oxidizer Turbine Inlet	POTI-1A	TG3	0 to 200	x					
115	Oxidizer Turbine Outlet	POTO-1A	TG4	0 to 100	x					
116	Oxidizer Turbine Outlet	POTO-2	TG4	0 to 100	x					
117	Bypass Nozzle Inlet	PGBNI		0 to 200	x					

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Pressure			psia						
118	PU Valve Inlet	PPUVI-1A	PO8	0 to 1000	x					
119	PU Valve Outlet	PPUVO-1A	PO9	0 to 500	x					
120	Fuel Pump Balance Piston Cavity	PFPC-1A		0 to 1000	x					
121	Oxidizer Pump Bearing Coolant	POPBC-1A		0 to 500	x					
122	Oxidizer Pump Bearing Coolant	POPBC-2		0 to 500	x					
123	Oxidizer Pump Primary Seal Cavity	POPSC-1A		0 to 50	x					
124	Fuel Start Tank	PFST-1P	TF1	0 to 1500	x		x			
125	Fuel Start Tank	PFST-2		0 to 1500						
126	He Tank	PHET-1P	NN1	0 to 3500	x		x			
127	He Tank	PHET-2	NN1	0 to 3500						
128	Engine Regulator Outlet	PHRO-1A		0 to 750	x					
129	Engine Regulator Outlet	PHRO-2		0 to 750	x					
130	Fuel Pump Interstage	PFPS-1P		0 to 200	x	x	x			
131	Fuel Recirculation Pump Return Line	PFRPR		0 to 50	x					
132	Oxidizer Recirculation Pump Return Line	PORPR		0 to 100	x					
133	Fuel Recirculation Pump Outlet	PFRPO		0 to 100	x					
134	Oxidizer Recirculation Pump Outlet	PORPO		0 to 200	x					
135	Fuel Tank Ullage	PFUT		0 to 100	x					
136	Oxidizer Tank Ullage	POUT		0 to 100	x					
137	He Supply	PHES		0 to 5000	x					
138	Fuel Tank Pressure Vent Line	PFVL		0 to 500	x					
139	Fuel Pump Inlet	PFPI-3		0 to 200		x	x			
140	Oxidizer Pump Inlet	POPI-3		0 to 200		x	x			

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	Pressure			psia						
141	Fuel Tapoff Orifice Outlet	PFOI-1A		0 to 1000	x					
142	Pneumatic Control Module Outlet	PHECMO		0 to 750	x					
143	Fuel Pump Inlet	PFPI-1		0 to 100	x					
144	Fuel Pump Inlet	PFPI-2		0 to 100	x					
145	Oxidizer Pump Inlet	POPI-1		0 to 100	x					
146	Oxidizer Pump Inlet	POPI-2		0 to 100	x					
147	Fuel Pump Discharge	PFPD-1P	PF3	0 to 1500	x					
148	Fuel Pump Discharge	PFPD-2	PF2	0 to 1500	x	x	x			
149	Oxidizer Pump Discharge	POPD-1P	PO3	0 to 1500	x					
150	Oxidizer Pump Discharge	POPD-2	PO2	0 to 1500	x	x	x			
151	Main Fuel Injector	PFJ-1A	CF2	0 to 1000	x		x			
152	Main Fuel Injector	PFJ-2	CF2A	0 to 1000	x	x				
153	Main Oxidizer Injector	POJ-1A	CO3	0 to 1000	x					
154	Main Oxidizer Injector	POJ-2	CO3A	0 to 1000	x	x	x			
155	Fuel Pump Balance Cavity	PFPC-2		0 to 1000	x					
156	Oxidizer Pump Primary Seal Cavity	POPSC-2		0 to 50	x					
157	Oxidizer Recirculation Pump	PHEOP		0 to 500	x					
158	Cell	PA1		0 to 0.5	x		x			
159	Cell	PA2		0 to 1.0	x	x				
160	Cell	PA3		0 to 5	x			x		
161	Spray Chamber	PEC48-1		0 to 1.0	x					
162	Thrust Chamber Fuel Jacket Purge	PTCFJP		0 to 100	x					
163	Main LO_2 Valve Closing Control	POVCC		0 to 500	x					Added on Test 18

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Charp	Event Recorder	Remarks
	Flow			gpm						
164	Main Fuel Flowmeter	QF-1A	PFF	0 to 9,000	x		x			
165	Main Fuel Flowmeter	QF-2	PFFA	0 to 9,000	x	x	x			
166	Main Oxidizer Flowmeter	QO-1A	POF	0 to 3,000	x		x			
167	Main Oxidizer Flowmeter	QO-2	POFA	0 to 3,000	x	x	x			
168	Fuel Recirculation Flowmeter	QFRP		0 to 160	x					
169	Oxidizer Recirculation Flowmeter	QORP		0 to 50	x					
170	Fuel Flow Stall Approach Monitor	QF-2SD		0 to 9,000	x		x			
	Speed			rpm						
171	Fuel Pump	NFP-1P	PFV	0 to 30,000	x		x			
172	Oxidizer Pump	NOP-1P	POV	0 to 12,000	x		x			
173	Fuel Childown Motor	NFRP		0 to 15,000	x					
174	Oxidizer Chilldown Motor	NORP		0 to 15,000	x					
	Event									
175	ES Command	EES			x		x			
176	Engine Cutoff Signal	EECO			x	x	x			
177	Fuel Prevalve Closed/Open	EFPVC/O			x		x			Added on Test 18
178	Oxidizer Prevalve Open	EOPVO			x		x			Added on Test 18
179	Oxidizer Prevalve Closed	EOPVC			x		x			Added on Test 18
180	He Control Solenoid On	EHCS			x		x			
181	Start Tank Discharge Control Solenoid On	ESTDCS			x	x	x			
182	Ignition Phase Control Solenoid On	EIPCS			x		x			
183	Ignition Detected	EID			x		x			

TABLE IV (Continued)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart	Event Recorder	Remarks
	<u>Event</u>									
184	Main-Stage Control Solenoid On	EMCS			x		x			
185	Engine Cutoff Lock-In	EECL			x		x			
186	MS OK No. 1 Pressurized	EMP-1			x		x			
187	MS OK No. 2 Pressurized	EMP-2			x		x			
188	Fuel Injection Temperature OK	EFJT			x		x			
	<u>Position</u>									
189	PU Valve Pot Output	LPUTOP			x		x	x		
190	Main Fuel Valve	LFVT			x		x			
191	MOV	LOVT			x		x			
192	GG Valve	LGGVT			x		x			
193	Oxidizer Turbine Bypass Valve	LOTBVT			x		x			
194	STDV	LSTDVT			x		x			
	<u>Vibration</u>			<u>g</u>						
195	Thrust Chamber Dome	UTCD-1		±500		x	x			
196	Thrust Chamber Dome	UTCD-2		±50		x	x			
197	Fuel Pump	UFPR		0 to 2000		x				
198	Oxidizer Pump	UOPR		0 to 2000		x				
199	Thrust Chamber Dome	UTCD-3		0 to 2000		x	x			
200	No. 1 VSC Counts	U1VSC					x			
201	No. 2 VSC Counts	U2VSC					x			

TABLE IV (Concluded)

Item No.	Parameter	AEDC Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	Event Recorder	Remarks
	Rate									
202	GG Spark Rate No. 1	RGGS-1					x			
203	GG Spark Rate No. 2	RGGS-2					x			
204	ASI Spark Rate No. 1	RASIS-1					x			
205	ASI Spark Rate No. 2	RASIS-2					x			
	Forces			lbf						
206	Side-Load Forces (Pitch)	FSP-1		±20,000	x		x	x		
207	Side-Load Forces (Yaw)	FSY-1		±20,000	x		x	x		
	Current									
208	Control Current	ICC			x		x			
209	Ignition Current	IIC			x		x			
	Volts									
210	Control Bus	VCB			x		x			
211	Ignition Bus	VIB			x		x			

TABLE V
ENGINE PURGE SEQUENCE AT AEDC

		Air On; Oxidizer Drop; Pre-conditioning; Engine Start; Engine Cutoff; Restart Test Engine Start; Engine Cutoff
Turbopump and GG (Purge Manifold System)	He, 82 - 125 psia 50 - 200°F at Customer Connect Panel, 6 scfm Nominal	10 min; 2-min Fuel; 1-3 min; Minimum Following Tank Pressurization*; 1-3 min; 10 min
Oxidizer Dome and GG Oxidizer Injector (Engine Pneumatic System)	He, 400 ± 25 psig at Engine Pneumatic Package Outlet, 50 to 200°F at He Fill Customer Connect, 230 scfm Nominal	15 min; 1 sec (Supplied by Engine He Tank during Start and Cutoff Transient
Oxidizer Dome (Facility Line to Instrumentation Port COA3)	N_2, 400 - 450 psig, 100 - 200°F at Facility Check Valve, 200 scfm Minimum	Duration of Hold; 10 min
Oxidizer Turbopump Intermediate Seal Cavity (Engine Pneumatic System)	He, 400 ± 25 psig at Engine Pneumatic Package Outlet, Engine Ambient Temperature, 2500 - 7000 scim	15 min; Main-Stage Operation (Supplied by Engine He Tank)
Thrust Chamber Jacket (Purge and Preconditioning through Customer Connect Line)	He, 40 - 60 psig, 50 - 200°F at Customer Connect Panel, 60 scfm Nominal	Any Time Water Is On†
	He, 12 - 14 psig 50 - 200°F at Customer Connect Panel, 10 scfm Nominal	Except When High Purge On‡; Duration of Hold‡; 10 min‡
	He, 1000 psig at Customer Connect Panel, 10 - 20 lb_m/min	

*Limitation imposed.

†When water is off, purge occurs only for 15 min after oxidizer drop.

‡Except when high purge on.

TABLE VI
TEST MATRIX

Run Number		12A	13B	15A	13B	14	15A	15B	15C	16	17A	17B	17C	18	18
Run Duration, sec		5.0	60.0	30.0	5.0	5.0	40.0	5.0	5.0	50.0	50	5.0	30.0	30.0	30.0
Fuel Pump Inlet Conditions at ES	Pressure, psia	$54 \pm ^{3}_{0}$	$37 \pm ^{0}_{5}$	$37 \pm ^{0}_{3}$	38.5 ± 1.5	$34 \pm ^{5}_{0}$	36.5 ± 1.5	38.5 ± 1.5	$37 \pm ^{0}_{3}$	$37 \pm ^{0}_{2}$	40 ± 1.5	40 ± 1.5	$30 \pm ^{3}_{0}$	$30 \pm ^{3}_{0}$	$30 \pm ^{3}_{0}$
	Temperature, °F	-421.5 ± 0.4	-420 ± 0.4	-420 ± 0.4	-431 ± 0.4	-431.6 ± 0.4	-421 ± 0.4	-421 ± 0.4	-420.1 ± 0.4	-423 ± 0.4	-421.8 ± 0.4	-421.8 ± 0.4	-432 ± 0.4	-422 ± 0.4	-431.3 ± 0.4
Oxidizer Pump Inlet Conditions at ES	Pressure, psia	$36 \pm ^{5}_{0}$	$36 \pm ^{3}_{0}$	$56 \pm ^{3}_{0}$	$58.5 \pm ^{3}_{0}$	$58 \pm ^{3}_{0}$	41.5 ± 1.5	41.5 ± 1.5	$38 \pm ^{3}_{0}$	$41 \pm ^{0}_{2}$	45 ± 1.5	45 ± 1.5	45 ± 1	$55 \pm ^{3}_{0}$	$55 \pm ^{3}_{0}$
	Temperature, °F	-290 ± 0.4	-290 ± 0.4	-298.4 ± 0.4	-289.2 ± 0.4	-280.1 ± 0.4	-291.1 ± 0.4	-291.1 ± 0.4	-286.5 ± 0.4	-295 ± 0.4	-285 ± 0.4	-283 ± 0.4	-293 ± 0.4	-297.4 ± 0.4	-298 ± 0.4
Start Tank Conditions at ES	Pressure, psia	$1250 \pm ^{25}_{0}$	$1575 \pm ^{0}_{25}$	$1375 \pm ^{25}_{0}$	$1350 \pm ^{0}_{35}$	$1250 \pm ^{25}_{0}$	$1350 \pm ^{25}_{0}$	$1350 \pm ^{25}_{0}$	$1375 \pm ^{25}_{0}$	$1350 \pm ^{25}_{0}$	1410 ± 10	1380 ± 10	1410 ± 10	1250 ± 10	1250 ± 10
	Temperature, °F	$-140 \pm ^{0}_{30}$	$-200 \pm ^{30}_{0}$	$-170 \pm ^{0}_{30}$	$-200 \pm ^{0}_{25}$	$-170 \pm ^{0}_{30}$	$-200 \pm ^{0}_{25}$	$-200 \pm ^{0}_{25}$	$-170 \pm ^{0}_{50}$	$-200 \pm ^{0}_{25}$	-250 ± 10	-225 ± 10	-250 ± 10	-140 ± 10	-140 ± 10
Thrust Chamber Temperature Conditions at ES, °F	Throat	-88 ± 10	-215 ± 10	-215 ± 10	---	-100 or Colder	---	---	-215 ± 10	---	-200 ± 10	---	-80 ± 10	---	-200 ± 10
	Exit	-165 ± 10	-225 ± 10	-225 ± 10	---	-100 or Colder	---	---	-225 ± 10	---	---	---	---	---	---
Fuel Lead Time, sec		3.0	5.0	1.0	4.5	1.0	4.5	4.5	3.0	8.0	3.5	8.0	2.5	8.0	2.5
Fuel in Engine Time, min		60	120	120	10	120	60	10	---	60	60	10	60	60	60
Oxidizer in Engine Time, min		60	120	120	10	130	60	10	---	60	60	10	60	60	60
Propellant Recirculation, min		10	15	15	10	10	10	10	15	10	10	10	10	10	10
PU Valve Position	ES, deg	0	0	0	0	0	0	-22	0	-32	0	-29	0	-29	0
	Main Stage, deg	0	0	33.3	---	0	33.3	---	0	33.3	33.3	-29	-22	35.3	35.3
	Excursion Time, sec	---	---	5	---	---	10	---	---	5	5	---	5	5	5
Prevalve Sequencing Logic		Normal	Normal	Normal	Normal	Normal	Normal	Normal	Normal	Normal	Auxiliary	Normal	Auxiliary	Normal	Auxiliary
MOV Closing Control Line Temperature, °F		---	---	---	---	---	---	---	---	---	---	---	---	$-100 \pm ^{50}_{0}$	$-75 \pm ^{0}_{25}$
Pneumatic Control Package Temperature, °F		---	---	---	---	---	---	---	---	---	---	---	---	---	$-75 \pm ^{0}_{25}$
Crossover Duct Temperature, °F		---	---	---	---	---	---	---	---	---	$-75 \pm ^{0}_{25}$	$125 \pm ^{0}_{35}$	$-75 \pm ^{0}_{35}$	$-50 \pm ^{25}_{0}$	$-75 \pm ^{0}_{35}$
Time between Multiple Tests, min		120	---	50	---	---	30	---	---	---	---	---	---	---	---

TABLE VII
SUMMARY OF TEST CONDITIONS AND RESULTS

Test Number		13A	12B	13A	13B	14	13A	13B	13C	18	17A	17B	17C	18	16
Test Date		12-2-66	13-3-66	13-13-66	12-13-66	13-20-66	1-4-67	1-4-67	1-4-67	1-13-67	1-17-67	1-19-67	1-19-67	1-38-67	3-3-67
Pressure Altitude at ES		86,000	108,000	100,000	108,000	103,000	106,000	111,000	110,000	111,000	93,000	108,000	107,000	99,000	105,000
Firing Duration, sec		5,075	50.0	30.07	1,333	5.07	40.07	1.38	3.87	30.07	30.07	5.07	30.07	30.07	30.07
Fuel Pump Inlet Conditions at ES	Pressure, psia	34.4	38.3	36.2	38.1	34.88	38.58	36.13	33.353	35.4	40.378	39.914	32.494	31.586	31.718
	Temperature, °F	-420.0	-430.00	-418.9	-421.3	-421.88	-420.8	-420.47	-420.33	-422.05	-422.44	-421.49	-421.97	-433.08	-431.03
Oxidizer Pump Inlet Conditions at ES	Pressure, psia	40.88	40.8	40.8	41.3	41.7	42.851	43.23	40.409	38.893	48.397	43.389	43.830	27.533	43.03
	Temperature, °F	-289.97	-289.79	-386.1	-291.2	-390.6	291.17	-380.88	-386.81	-293.74	-393.19	-293.34	-293.03	-296.35	-396.03
Start Tank Conditions at ES	Pressure, psia	1,259	1,368	1,364	1,359	1,289	1,363	1,361	1,370	1,359	1,388	1,372	1,385	1,349	1,340
	Temperature, °F	-161.39	-190.4	-192.8	-218.4	-146.2	-214	-219	-187.88	-213.5	-210	-330	-201	-148	-148
He Tank Conditions at ES	Pressure, psia	2,901	3,828	3,683	2,811.6	3,082	3,128	2,032	2,870	3,033	3,125	3,118	3,898	2,717	3,045
	Temperature, °F	-144.41	-183.8	-188.9	-214.8	-133.8	-213	-318	-197.15	-215.16	-219	-237	-213	-155	-130
Thrust Chamber Temperature Conditions at ES, °F	Throat	-110	-215	-210	-1	-181	82	-1	-234	43	-311	52	-88	68	-331
	Exit	-160	-319	-207	-8	-107	48	-2	-333	48	-138	13	-94	58	-130
Fuel Lead Time, sec		3.110	3.093	1.038	4.720	1.951	4.438	4.455	3.114	8.070	2.648	8.308	2.645	7.974	3.333
Fuel in Engine Time, min		63	155+	99+	10	108	78	10	40+	145+	122	10	60	35+	60
Oxidizer in Engine Time, min		43	153+	99+	10	108	78	10	40+	145+	122	10	60	35+	60
Propellant Recirculation, min		12	18	15	10	10	10	10	17	11	10	8	11	10	10
VSC Duration, msec		40	18	80	40	15	0	25	30	21	35	8	30	15	8
GG Temperature, °F	Initial Spike	1,783	3,080	3,161	3,172	1,750	1,587	1,880	2,071	1,763	2,026	2,092	1,960	1,560	1,984
	Second Spike	1,604	---	---	3,426	1,808	1,799	2,132	---	---	1,740	3,118	1,887	---	---
Thrust Chamber Dome Prime Time, sec		0.890	0.994	0.071	0.818	1.003	0.984	0.831	0.983	0.898	0.974	0.945	0.963	1.037	1.007
Time from STDV to 2nd Stage MOV Ramp, msec		1,183	1,103	0.894	---	1,025	1,201	1,261	1,186	1,013	1,146	1,242	1,188	1,080	1,110
Main-Stage OK Signal, sec		1.780	1.750	1.861	---	1.713	1.732	---	1.718	1.902	1.731	1.721	1.880	1.070	1.718
Time to p_c = 350 psia-sec		3.032	2.063	1.890	---	1.060	1.950	---	1.999	2.311	1.983	3.116	1.923	3.385	1.880
Crossover Duct Temperature, °F	TFTD-3	44	35	23	203	45	33	212	33	-33	-52	160	-45	-41	-58
	TFTD-4	43	35	23	211	48	34	218	38	-38	-83	151	-33	43	-48
PU Valve Position, deg	ES	-0.7	-0.7	-0.8	-0.8	-0.8	-0.5	-22.8	-0.8	-33.8	-0.7	-27.9	-0.6	-27.9	-0.6
	Main Stage	-0.7	-0.7	33.8	---	-0.0	33.8	---	-0.6	32.7	32.6	-27.9	-22.9	32.6	32.7
	Excursion Time	---	---	8.025	---	---	16.0	---	---	18.0	3.8	---	3.8	8.0	35.3
Prevalve Sequencing Logic		Normal	Normal	Normal	Normal	Normal	Normal	Normal	Normal	Normal	Aux	Normal	Aux	Normal	Aux
Time between Multiple Tests, min		180	---	31	---	---	30	183	---	---	58	154	---	---	---
MOV Closing Control Line Temperature, °F		---	---	---	---	---	---	---	---	7.24	-1.26	19.34	-7.6	14.112	-68.5
Pneumatic Control Package Temperature, °F		---	---	---	---	---	---	---	---	---	---	---	---	---	-66.25

TABLE VIII

ENGINE START AND SHUTDOWN SEQUENCE

Start																	
Test Number	Main Fuel Valve Open: Valve Delay Time, sec	Main Fuel Valve Open: Valve Opening Time, sec	STDV Open: Valve Delay Time, sec	STDV Open: Valve Opening Time, sec	STDV Closed: Valve Delay Time, sec	STDV Closed: Valve Closing Time, sec	MOV, First Stage Open: Valve Delay Time, sec	MOV, First Stage Open: Valve Opening Time, sec	MOV, Second Stage Open: Valve Delay Time, sec	MOV, Second Stage Open: Valve Opening Time, sec	GG Fuel Poppet Open: Valve Delay Time, sec	GG Fuel Poppet Open: Valve Opening Time, sec	GG Oxidizer Poppet Open: Valve Delay Time, sec	GG Oxidizer Poppet Open: Valve Opening Time, sec	Oxidizer Turbine Bypass Valve Closed: Valve Delay Time, sec	Oxidizer Turbine Bypass Valve Closed: Valve Closing Time, sec	Ignition Phase Timer, sec
12A	0.050	0.069	0.153	0.136	0.097	0.262	0.050	0.052	0.740	2.070	0.076	0.056	0.168	0.078	0.210	0.268	0.445
12B	0.050	0.074	0.166	0.150	0.100	0.268	0.052	0.057	0.750	2.020	0.079	0.056	0.180	0.072	0.220	0.267	0.436
13A	0.050	0.064	0.152	0.148	0.097	0.268	0.052	0.054	0.556	2.140	0.080	0.048	0.170	0.064	0.216	0.269	0.438
*13B	0.050	0.064	0.166	0.150	0.099	0.268	0.050	0.055	---	---	0.081	0.052	0.182	0.076	0.201	0.289	0.438
14	0.050	0.067	0.149	0.134	0.097	0.263	0.053	0.057	0.585	2.170	0.086	0.044	0.172	0.061	0.208	0.271	0.440
15A	0.050	0.070	0.164	0.147	0.098	0.267	0.053	0.054	0.762	2.056	0.078	0.052	0.177	0.064	0.215	0.265	0.439
*15B	0.051	0.060	0.167	0.146	0.098	0.266	0.050	0.054	0.814	0.424*	0.080	0.051	0.182	0.075	0.205	0.295	0.439
15C	0.050	0.067	0.166	0.151	0.097	0.276	0.052	0.057	0.747	2.086	0.085	0.049	0.178	0.070	0.205	0.277	0.438
16	0.051	0.064	0.160	0.144	0.094	0.257	0.051	0.055	0.574	2.174	0.080	0.050	0.170	0.064	0.215	0.255	0.439
17A	0.050	0.070	0.164	0.153	0.095	0.271	0.052	0.053	0.708	2.094	0.080	0.053	0.179	0.063	0.214	0.271	0.438
17B	0.047	0.071	0.172	0.158	0.099	0.270	0.049	0.052	0.803	1.970	0.079	0.055	0.182	0.083	0.200	0.565	0.439
17C	0.047	0.070	0.166	0.154	0.098	0.270	0.050	0.053	0.762	2.053	0.080	0.051	0.181	0.068	0.205	0.280	0.437
18	0.049	0.067	0.148	0.137	0.096	0.270	0.050	0.056	0.608	2.174	0.078	0.049	0.167	0.070	0.190	0.282	0.442
19	0.050	0.062	0.144	0.131	0.092	0.258	0.052	0.048	0.678	2.102	0.077	0.052	0.174	0.073	0.220	0.258	0.432

*Premature Cutoff

TABLE VIII (Concluded)

Test Number	Shutdown									
	MOV Closed		Main Fuel Valve Closed		GG Oxidizer Poppet Closed		GG Fuel Poppet Closed		Oxidizer Turbine Bypass Valve Open	
	Valve Delay Time, sec	Valve Closing Time, sec	Valve Delay Time, sec	Valve Closing Time, sec	Valve Delay Time, sec	Valve Closing Time, sec	Valve Delay Time, sec	Valve Closing Time, sec	Valve Delay Time, sec	Valve Opening Time, sec
12A	0.057	0.173	0.124	0.295	0.020	0.028	0.080	0.018	0.210	0.270
12B	0.083	0.189	0.118	0.329	0.020	0.030	0.080	0.038	0.250	0.618
13A	0.083	0.192	0.129	0.333	0.030	0.014	0.072	0.014	0.260	0.573
*13B	0.037	0.050	0.110	0.307	0.035	0.014	0.085	0.016	0.169	0.558
14	0.058	0.189	0.126	0.328	0.033	0.018	0.078	0.018	0.227	0.541
15A	0.077	0.181	0.117	0.298	0.020	0.026	0.076	0.031	0.268	0.626
*15B	0.020	0.068	0.108	0.292	0.038	0.015	0.086	0.020	0.180	0.562
15C	0.058	0.192	0.120	0.334	0.034	0.016	0.066	0.024	0.250	0.515
16	0.082	0.193	0.124	0.318	0.030	0.015	0.071	0.026	0.306	0.578
17A	0.079	0.194	0.130	0.333	0.017	0.029	0.076	0.024	0.308	0.560
17B	0.054	0.175	0.113	0.297	0.020	0.030	0.075	0.023	0.225	0.520
17C	0.079	0.177	0.120	0.310	0.025	0.028	0.080	0.023	0.268	0.536
18	0.079	0.195	0.127	0.325	0.031	0.014	0.073	0.020	0.262	0.594
19	0.078	0.172	0.120	0.310	0.032	0.015	0.070	0.030	0.263	0.603

*Premature Cutoff

TABLE IX
MAIN OXIDIZER VALVE FINAL PRE-TEST SEQUENCE CHECKS

	First Stage		Second Stage	
Test	Valve Delay	Valve Travel Time	Valve Delay	Valve Travel Time
12	45	44	572	1766
13	43	47	580	1810
14	42	44	586	1795
15	47	44	600	1795
16	45	45	584	1802
17	48	46	600	1839
18	45	43	580	1810
19	44	47	585	1803
Average	45	45	586	1803
Spec.	50 ± 20	50 ± 25	580 ± 75	1825 ± 75

TABLE X
MAXIMUM OXIDIZER PUMP SPIN SPEEDS RESULTING FROM START TANK BLOWDOWN

Test	N_O, rpm	ΔN_O*, rpm
12A	3367	218
12B	3490	368
13A	3446	212
13B	3976	34
14	3289	101
15A	3550	75
15B	3833	133
15C	3461	355
16	3298	253
17A	3437	276
17B	3615	186
17C	3531	137
18	3020	237
19	3180	257

*Oxidizer pump maximum spin speed decay before engine acceleration to main stage.

TABLE XI
ENGINE START CONDITIONS, FLIGHT AND AEDC

Run No.	Start Tank		Thrust Chamber Temperature, °F		Oxidizer Pump Inlet		Fuel Pump Inlet	
	Temperature, °F	Pressure, psia	Throat	Exit	Pressure, psia	Temperature, °F	Pressure, psia	Temperature, °F
			Throat	Exit				
AS 201	-275	1273	-212	-245	40	-295	41	-421
AS 202	-171	1273	-222	-236	40.6	-295	39.3	-422
AS 203	-220	1300	-205	-240	42	-296	37	-421
17A	-210	1386	-211	-138	46.3	-293.2	40.4	-422
19	-146	1249	-231	-120	45	-295	31.8	-421

Run No.	Engine Area Ambient Temperature, °F	Fuel Turbine Outlet Temperature, °F	MOV Actuator Line Temperature, °F	MOV Closing Control Line Temperature, °F
AS 201	-40			
AS 202	-72		-135	-25
AS 203	-60	-29	-125	-25
17A	25	-50		-1
19	48	-43		-68

TABLE XII
ENGINE PERFORMANCE SUMMARY

Test Number		12B			13A	15A		16A	17A	17C	18A	19A	315001*	315001*
Data Slice Time, sec		29.5	39.5	49.5	29.5	29.5	39.5	29.5	29.5	28.5	29.5	29.5	80.0	78.9
Pressure Altitude, ft		101,000	100,000	100,000	94,000	87,000	87,000	99,000	93,000	101,000	96,000	96,000	84,000	87,000
Engine Performance	Thrust, lb_f	189,343.8	188,956.7	186,646.9	219,857.1	218,136.2	218,083.7	214,977.6	219,818.3	174,604.8	222,538.5	222,045	221,263	196,025
	Thrust, lb_f Vacuum	190,048.9	189,689.2	188,413.0	220,845.1	218,996.7	218,949.7	215,765.6	220,860.8	175,325.4	223,540.7	222,981	---	---
	Specific Impulse, lb_f-sec/lb_m	419.9	418.8	421.4	417.1	417.5	418.1	410.8	418.1	420.6	418.8	417.6	418.6	422.9
	Specific Impulse, lb_f-sec/lb_m Vacuum	421.4	420.5	423.1	419.0	419.1	417.9	412.1	420.1	422.4	420.8	419.2	---	---
	Mixture Ratio	5.043	5.043	5.000	5.631	5.626	5.652	5.888	5.618	4.760	5.707	5.704	5.510	4.894
	Fuel Weight Flow, lb_m/sec	74.63	74.68	74.61	78.49	78.86	78.79	78.17	79.48	72.07	79.24	79.33	81.20	78.55
	Oxidizer Weight Flow, lb_m/sec	376.35	376.50	373.06	447.58	443.68	445.31	445.44	446.27	343.04	452.24	452.46	447.39	384.02
	Total Weight Flow, lb_m/sec	450.87	451.18	447.67	527.08	522.51	524.09	523.61	525.70	415.11	531.48	531.78	528.59	463.57
Thrust Chamber Performance	Specific Impulse, lb_f-sec/lb_m	425.7	424.7	427.3	432.7	423.1	421.7	416.0	423.6	426.7	424.5	423.1	424.2	428.8
	Characteristic Velocity, ft/sec	7956.6	7938.8	7890.7	7838.4	7845.1	7818.9	7718.5	7859.8	8012.1	7862.3	7837.8	7946.7	8085.2
	Thrust Coefficient	1.7214	1.7209	1.7205	1.7351	1.7352	1.7351	1.7341	1.7347	1.7133	1.7372	1.737	1.718	1.704
	Thrust Coefficient Vacuum	1.7278	1.7278	1.7275	1.7429	1.7420	1.7420	1.7405	1.7429	1.7204	1.7442	1.7439	---	---
	Mixture Ratio	5.240	5.240	5.185	5.853	5.851	5.896	5.929	5.842	4.948	5.934	5.931	5.726	5.082
	Fuel Weight Flow, lb_m/sec	71.28	71.31	71.37	75.90	75.26	75.22	74.58	75.82	68.80	75.83	75.72	77.55	75.18
	Oxidizer Weight Flow, lb_m/sec	373.50	373.65	370.22	444.20	440.33	441.88	442.19	442.90	340.44	448.83	449.03	444.02	381.98
	Total Weight Flow, lb_m/sec	444.78	444.88	441.49	520.10	515.57	517.20	516.77	518.72	408.23	524.48	524.75	521.57	457.12
	Chamber Pressure, psia	645.8	644.5	643.8	743.6	737.9	737.8	727.7	743.6	598.2	753.3	750.4	756.2	675.1
Fuel Pump Performance	Inlet Stagnation Pressure, psia	34.8	35.3	35.2	34.3	33.7	33.5	33.8	37.8	30.9	31.1	30.6	38.4	36.5
	Inlet Density, lb_m/ft^3	4.30	4.30	4.30	4.32	4.37	4.37	4.41	4.41	4.39	4.41	4.38	4.40	4.40
	Discharge Stagnation Pressure, psia	1046.5	1046.9	1046.3	1200.2	1177.8	1178.7	1160.2	1191.9	967.6	1202.5	1221.3	1222.5	1102.6
	Discharge Density, lb_m/ft^3	4.43	4.43	4.42	4.45	4.47	4.47	4.50	4.51	4.47	4.51	4.49	4.52	4.51
	Head, ft	32,603.2	32,607.7	32,598.1	37,280.9	36,346.0	36,286.2	35,404.8	38,213.3	29,698.6	38,723.5	37,581.3	37,213.1	33,546.2
	Speed, rpm	25,133.8	25,158.1	25,106.5	26,780.2	26,383.6	26,410.6	25,980.6	26,462.8	24,135.0	26,514.8	28,719.1	28,829	25,690
	Weight Flow, lb_m/sec	75.46	75.52	75.48	80.37	78.76	79.69	79.06	80.34	72.88	80.18	80.23	81.97	79.39
	Efficiency	0.746	0.745	0.744	0.735	0.716	0.717	0.721	0.719	0.720	0.719	0.722	0.735	0.736
Oxidizer Pump Performance	Inlet Stagnation Pressure, psia	40.8	40.7	40.4	38.8	43.7	44.2	41.3	45.0	44.5	40.1	45.4	45.0	45.2
	Inlet Density, lb_m/ft^3	69.94	68.93	69.93	71.05	70.16	70.13	70.83	70.50	70.46	71.07	71.09	70.67	70.57
	Discharge Stagnation Pressure, psia	878.0	874.6	872.2	1082.1	1049.1	1050.4	1032.4	1085.4	795.5	1075.6	1075.4	1077.7	921.1
	Discharge Density, lb_m/ft^3	68.93	69.92	69.01	71.25	70.35	70.35	71.11	70.71	70.37	71.28	71.31	70.88	70.48
	Head, ft	1720.0	1717.8	1713.3	2065.8	2057.7	2059.5	2008.8	2077.7	1537.0	2091.6	2079.6	2103.9	1789.7
	Speed, rpm	7845.6	7846.2	7832.6	8496.0	8505.1	8511.8	8443.8	8520.4	7319.2	8540.0	8516.6	8580.1	7985.5
	Weight Flow, lb_m/sec	453.55	453.14	449.17	455.17	451.20	452.65	453.02	453.83	472.10	459.83	460.05	458.73	485.52
	Efficiency	0.810	0.810	0.810	0.801	0.802	0.802	0.802	0.802	0.807	0.802	0.802	0.802	0.810

*Data from J-2 2052 Log Book

TABLE XII (Concluded)

Test Number			12B			13A	15A		16A	17A	17C	18A	19A	315001*	316001*
Data Slice Time, sec			29.5	39.5	49.5	29.5	29.5	39.5	29.5	29.5	29.8	29.3	29.3	60.0	79.9
GG Performance	Chamber Pressure, psia		577.2/568.6	577.9/569.2	575.5/566.9	663.6/655.1	657.6/650.6	655.7/647.5	646.3/636.0	663.6/653.9	526.2/520.9	670.9/662.2	671.6/663.2	557.7/569.4	591.7/600.7
	Mixture Ratio, O/F		0.852	0.851	0.848	0.843	0.929	0.921	0.905	0.922	0.795	0.847	0.848	0.922	0.849
	Fuel Weight Flow, lb_m/sec		3.35	3.35	3.35	3.58	3.59	3.57	3.69	3.61	3.27	3.81	3.81	3.68	3.49
	Oxidizer Weight Flow, lb_m/sec		2.85	2.85	2.84	3.26	3.34	3.32	3.35	3.37	2.60	3.42	3.42	3.27	2.96
	Total Weight Flow, lb_m/sec		6.20	6.21	6.19	6.96	6.93	6.90	6.65	6.98	5.87	7.03	7.03	7.02	6.46
Fuel Turbine Performance	Weight Flow, lb_m/sec		6.20	6.21	6.19	6.86	6.93	6.90	6.65	6.98	5.87	7.03	7.03	7.02	6.45
	Inlet	Pressure, psia	626.5	527.2	624.9	618.2	614.0	611.0	602.0	619.0	499.0	625.0	626.0	620.6	556.2
		Temperature, °F	1051.4	1049.6	1044.3	1204.9	1182.6	1186.2	1140.6	1186.3	933.8	1213.4	1214.7	1180.4	1046.2
	Discharge	Pressure, psia	74.79	74.86	74.62	85.67	84.49	84.66	82.52	86.27	70.02	87.02	86.41	86.53	77.78
		Temperature, °F	652.9	655.6	651.9	762.4	745.0	762.4	725.6	759.1	589.6	777.1	759.2	749.8	653.8
	Pressure Ratio		7.38	7.36	7.36	7.40	7.46	7.41	7.40	7.36	7.21	7.37	7.42	7.36	7.37
	Developed Horsepower		5985	6006.0	6016.1	7416.4	7319.2	7333.3	7080.7	7260.6	6462.4	7444.2	7597.8	7644.6	6579.7
	Efficiency		0.651	0.551	0.655	0.575	0.572	0.577	0.568	0.573	0.551	0.571	0.580	0.568	0.561
Oxidizer Turbine Performance	Weight Flow, lb_m/sec		5.62	5.52	6.61	6.20	6.19	6.15	6.11	6.23	5.22	6.26	6.27	6.26	5.76
	Inlet	Pressure, psia	72.4	72.5	72.2	82.9	81.8	81.6	80.8	82.6	67.8	84.3	83.7	83.3	75.6
		Temperature, °F	652.9	655.0	651.9	762.4	745.0	752.4	752.8	759.1	368.8	777.1	739.2	740.1	651.6
	Discharge	Pressure, psia	28.08	28.10	26.04	32.27	31.60	31.69	31.15	32.41	26.22	32.6	32.33	31.95	26.71
		Temperature, °R	521.3	525.6	524.6	610.3	696.6	605.1	574.0	605.0	466.9	621.1	691.4	587.4	516.0
	Pressure Ratio		2.62	2.82	2.62	2.61	2.64	2.62	2.64	2.82	2.82	2.82	2.62	2.67	2.87
	Developed Horsepower		1750.4	1746.6	1727.6	2122.5	2105.6	2114.0	2061.3	2136.8	1833.3	2180.5	2168.0	2179.3	1869.2
	Efficiency		0.445	0.442	0.440	0.452	0.457	0.460	0.455	0.459	0.450	0.461	0.464	0.461	0.446
Propellant Tank Pressurant Flow	Oxidizer, lb/sec		0.6	0.0	0.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.85	1.62
	Fuel, lb/sec		0.86	0.86	0.86	0.89	0.91	0.90	0.90	0.91	0.82	0.92	0.96	6.77	0.74
PU Valve	Angle, deg		-0.7	-0.7	-0.6	32.6	32.6	32.6	32.7	32.6	-23.9	32.6	32.7	32.1	0.0
	Hydraulic Resistance, $sec^2/ft^3\text{-}in.^2$		5.74	5.79	5.80	1000.52	1000.52	1000.32	1000.52	1000.52	1.26	1000.52	1000.52	1000.64	6.27
	Weight Flow, lb_m/sec		77.2	76.6	76.1	7.6	7.5	7.5	7.6	7.6	129.1	7.5	7.6	7.5	79.0
Supplementary Engine Data	Oxidizer Pump Delta Speed, rpm		-29.08	-35.19	-45.21	-121.87	-137.75	-135.05	-171.11	-119.04	241.68	-107.52	-105.41	-107.14	2.09
	Fuel Pump Delta Speed, rpm		-550.60	-561.67	-610.66	-377.87	-596.46	-622.77	-558.02	-714.62	-755.46	-683.89	-617.11	-626.94	-607.87
	Fuel Turbine Map Power Ratio		0.9748	0.9748	0.9603	0.9964	0.9891	1.0070	0.9956	0.9988	0.9880	0.9973	1.0098	1.0152	1.0159
	Oxidizer Turbine Map Power Ratio		1.0114	1.0082	1.0023	1.0126	1.0084	1.0112	1.0044	1.0041	1.0541	1.0104	1.0153	1.072	1.0186

*Data from J-2 2052 Log Book

TABLE XIII
ENGINE NORMALIZED PERFORMANCE SUMMARY

Test Number		12B			13A	15A		16A	17A	17C	18A	19A	315001*	315001*
Data Slice Time, sec		29.5	39.5	49.5	29.5	28.5	39.5	29.5	29.5	29.5	29.5	29.5	60	79.9
Engine Performance	Thrust, lb_f	190,011.1	189,993.0	189,426.5	218,725.5	217,789.0	217,751.7	212,932.9	218,077.1	174,039.0	220,508.2	219,370	222,861	195,554
	Mixture Ratio, O/F	4.985	4.085	4.843	5.535	5.578	5.602	5.552	5.598	4.728	5.654	5.509	5.502	4.881
	Total Weight Flow, lb/sec	450.44	450.63	447.31	520.98	519.13	520.67	516.05	518.92	411.90	523.75	522.26	523.28	457.25
Thrust Chamber Performance	Chamber Pressure, psia	646.1	645.0	644.2	737.8	734.4	734.3	718.3	734.7	594.1	742.6	739.3	748.7	666.1
	Mixture Ratio, O/F	5.180	5.181	5.137	5.754	5.801	5.824	5.881	5.821	4.915	5.879	5.833	5.718	5.050
	Weight Flow, lb_m/sec	444.20	444.38	441.08	514.03	512.21	513.78	509.27	512.01	406.05	516.70	515.29	516.31	450.86
	Characteristic Velocity, ft/sec	7972.7	7955.6	8005.7	7865.2	7858.6	7833.2	7730.6	7864.9	8020.1	7876.5	7864.4	7947.8	8097.5
Fuel Pump Performance	Pump Speed, rpm	24,894.7	24,811.8	24,955.2	26,462.5	26,245.8	26,271.5	25,856.4	26,353.0	24,051.0	26,409.4	26,487.9	26,715	25,544
	Weight Flow, lb/sec	76.06	76.09	76.06	80.52	79.71	79.67	78.38	79.45	72.72	79.51	79.82	81.29	78.55
	Efficiency	0.746	0.745	0.743	0.734	0.719	0.717	0.721	0.718	0.720	0.719	0.721	0.735	0.736
Oxidizer Pump Performance	Pump Speed, rpm	7783.9	7793.7	7770.6	8454.0	8435.5	8442.5	8380.5	8443.0	7292.5	8482.4	8458.9	8498.7	7891.7
	Weight Flow, lb/sec	453.93	453.52	449.77	450.53	449.59	451.17	447.84	449.64	466.69	454.4	452.61	452.07	459.92
	Efficiency	0.810	0.810	0.810	0.801	0.801	0.801	0.801	0.801	0.808	0.802	0.802	0.801	0.810
Fuel Turbine Performance	Efficiency	0.550	0.550	0.554	0.573	0.571	0.575	0.568	0.572	0.551	0.571	0.579	0.587	0.580
	Pressure Ratio	7.36	7.35	7.35	7.40	7.46	7.41	7.40	7.36	7.31	7.37	7.43	7.35	7.36
	Inlet Temperature, °F	1030.3	1028.3	1023.4	1172.4	1169.5	1172.6	1128.5	1181.3	944.9	1189.1	1188.2	1152.8	1037.2
	Weight Flow, lb_m/sec	6.23	6.24	6.22	6.94	6.92	6.99	6.78	6.91	5.85	6.05	6.96	6.97	6.39
Oxidizer Turbine Performance	Efficiency	0.443	0.441	0.438	0.462	0.455	0.458	0.453	0.457	0.448	0.459	0.463	0.459	0.445
	Pressure Ratio	2.62	2.62	2.62	2.61	2.64	2.63	2.64	2.62	2.63	2.65	2.63	2.67	2.67
	Inlet Temperature, °F	638.3	640.2	637.3	739.3	735.9	742.8	716.8	754.4	583.5	767.5	740.1	740.5	646.7
	Weight Flow, lb_m/sec	5.55	5.56	5.55	6.19	6.18	6.14	6.04	6.16	5.21	6.20	6.21	6.21	5.69
GG Performance	Mixture Ratio, O/F	0.840	0.839	0.839	0.923	0.922	0.923	0.897	0.929	0.790	0.939	0.933	0.918	0.844
	Chamber Pressure, psia	569.8	570.5	568.2	650.3	648.1	645.0	630.2	648.6	525.8	654.1	654.0	652.1	584.9

*Data from J-2 2052 Log Book

APPENDIX III
TEST MEASUREMENTS REQUIRED BY PERFORMANCE PROGRAM

Thrust Chamber (Injector Face) Pressure, psia

Thrust Chamber Fuel and Oxidizer Injection Pressure, psia

Thrust Chamber Fuel Injection Temperature, °F

Fuel and Oxidizer Flowmeter Speeds, cps

Fuel and Oxidizer Engine Inlet Pressures, psia

Fuel and Oxidizer Pump Discharge Pressures, psia

Fuel and Oxidizer Engine Inlet Temperatures, °F

Fuel and Oxidizer (Main Valves) Temperatures, °F

PU Valve Center Tap Voltage, volts

PU Valve Position, volts

Fuel and Oxidizer Pump Speeds, rpm

GG Chamber Pressure, psia

GG (Bootstrap Line at Bleed Valve) Temperature, °F

Fuel* and Oxidizer Turbine Inlet Pressure, psia

Oxidizer Turbine Discharge Pressure, psia

Fuel and Oxidizer Turbine Inlet Temperature, °F

Oxidizer Turbine Discharge Temperature, °F

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*At AEDC, fuel turbine inlet pressure is estimated from GG chamber pressure.

NOMENCLATURE

A	Area, in.^{2}
B	Horsepower
C	Coefficient
C*	Characteristic velocity, ft/sec
D	Diameter, in.
F	Thrust, lb_f
H	Head, ft
h	Enthalpy, Btu/lb_m
I	Impulse
M	Molecular weight
N	Speed, rpm
P	Pressure, psia
Q	Flow rate, gpm
R	Resistance, sec^2/ft^3-in.^{2}
r	Mixture ratio, O/F
T	Temperature, °F
TC*	Theoretical characteristic velocity, ft/sec
W	Weight flow, lb/sec
Z	Differential pressure, psi
β	Ratio
γ	Ratio of specific heats
η	Efficiencies
θ	Degrees
ρ	Density, lb/ft^3

SUBSCRIPTS

A	Ambient
AA	Ambient at thrust chamber exit
B	Bypass nozzle

BIR	Bypass nozzle inlet (Rankine)
BNI	Bypass nozzle inlet (total)
C	Thrust chamber
CF	Thrust chamber, fuel
CO	Thrust chamber, oxidizer
CV	Thrust chamber, vacuum
E	Engine
EF	Engine fuel
EM	Engine measured
EO	Engine oxidizer
EV	Engine, vacuum
e	Exit
em	Exit measured
F	Thrust
FM	Fuel measured
FTI	Fuel turbine inlet
FV	Thrust, vacuum
f	Fuel
G	Gas generator
GF	GG fuel
GO	GG oxidizer
H1	Hot gas duct No. 1
H1R	Hot gas duct No. 1 (Rankine)
H2R	Hot gas duct No. 2 (Rankine)
IF	Inlet fuel
IO	Inlet oxidizer
ITF	Isentropic turbine fuel
ITO	Isentropic turbine oxidizer
N	Nozzle
NB	Bypass nozzle (throat)

NV	Nozzle, vacuum
O	Oxidizer
OC	Oxidizer pump calculated
OF	Outlet fuel pump
OFIS	Outlet fuel pump isentropic
OM	Oxidizer measured
OO	Oxidizer outlet
PF	Pump fuel
PO	Pump oxidizer
PUVO	PU valve oxidizer
RNC	Ratio bypass nozzle, critical
SC	Specific, thrust chamber
SCV	Specific thrust chamber, vacuum
SE	Specific, engine
SEV	Specific, engine vacuum
T	Total
TEF	Turbine exit fuel
TEFS	Turbine exit fuel (static)
TF	Fuel turbine
TIF	Turbine inlet fuel (total)
TIFM	Turbine inlet, fuel, measured
TIFS	Turbine inlet fuel isentropic
TIO	Turbine inlet oxidizer
TO	Turbine oxidizer
t	Throat
V	Vacuum
v	Valve
XF	Fuel tank repressurant
XO	Oxidizer tank repressurant

PERFORMANCE PROGRAM EQUATIONS

THRUST

Thrust Chamber, Vacuum

$$F_{CV} = C\,(P_c)^2 + B\,(P_c) + A$$

Empirical Determination from Curve Fit of Thrust versus P_c

Thrust Chamber

$$F_C = F_{CV} - P_{AA}\,A_e$$

$$A_e = A_{em} + 12.8$$

$$P_A = \text{Measured Cell Pressure}$$

Engine, Vacuum

$$F_{EV} = F_{CV}$$

Engine

$$F_E = F_C$$

MIXTURE RATIO

Engine

$$r_E = \frac{W_{EO}}{W_{EF}}$$

$$W_{EO} = W_{OM} - W_{XO}$$

$$W_{EF} = W_{FM} - W_{XF}$$

Thrust Chamber

$$r_C = \frac{W_{CO}}{W_{CF}}$$

$$W_{CO} = W_{OM} - W_{XO} - W_{GO}$$

$$W_{CF} = W_{FM} - W_{XF} - W_{GF}$$

$$W_{XO} = \text{Standard } 0.8 \text{ lb/sec}$$

$$W_{XF} = \text{Standard } 1.8 \text{ lb/sec}$$

$$W_{GO} = W_T - W_{GF}$$

$$W_{GF} = W_T\,(1 + r_G)$$

$$W_T = \frac{P_{TIF}\,A_{TIF}\,K_7}{TC^*_{TIF}}$$

$$K_7 = 32.174$$

Normalized engine and thrust chamber vacuum data calculated as measured, except all flows are normalized using standard inlet pressures, temperatures and densities listed below:

$$P_{IO}\ STD = 39\ \text{psia}$$
$$P_{IF}\ STD = 30\ \text{psia}$$
$$\rho_{IO}\ STD = 70.79\ \text{lb/ft}^3$$
$$\rho_{IF}\ STD = 4.40\ \text{lb/ft}^3$$
$$T_{IO}\ STD = -295.2°F$$
$$T_{IF}\ STD = 422.5°F$$

SPECIFIC IMPULSE

Engine

$$I_{SE} = \frac{F_E}{W_E}$$

$$W_E = W_{EO} + W_{EF}$$

Engine, vacuum

$$I_{SEV} + \frac{F_{EV}}{W_{EV}}$$

$W_{EV} = W_E$ Normalized using standard inlet pressures, temperatures, and densities

Chamber

$$I_{SC} = \frac{F_C}{W_C}$$

$$W_C = W_{CO} + W_{CF}$$

Chamber, vacuum

$$I_{SCV} = \frac{F_{CV}}{W_C}$$

$W_{CV} = W_C$ Normalized using standard inlet pressures, temperatures, and densities

CHARACTERISTIC VELOCITY

Thrust Chamber

$$C^* = \frac{K_7\ P_c\ A_t}{W_C}$$

$$K_7 = 32.174$$

Thrust Chamber, Vacuum

$$C^*_V = \frac{K_7\ P_{CV}\ A_t}{W_{CV}}$$

$$K_7 = 32.174$$

Nozzle

$$C_N^* = \frac{C^*}{K_6}$$

$$K_6 = 1.086$$

Nozzle, Vacuum

$$C_{NV}^* = \frac{C_V^*}{K_6}$$

$$K_6 = 1.086$$

THRUST COEFFICIENT

Engine

$$C_F = \frac{F_C}{P_c A_t}$$

Engine, Vacuum

$$C_{FV} = \frac{F_{CV}}{P_c A_t}$$

DEVELOPED PUMP HEAD

Oxidizer

$$H_O = K_4 \left(\frac{P_{OO}}{\rho_{OO}} - \frac{P_{IO}}{\rho_{IO}}\right)$$

$$K_4 = 144$$

$$\rho = \text{NBS Values } f(P,T)$$

Fuel

$$H_f = 778.16\ \Delta h_{OFIS}$$

$$\Delta h_{OFIS} = h_{OFIS} - h_{IF}$$

$$h_{OFIS} = f(P,T)$$

$$h_{IF} = f(P,T)$$

Fuel and Oxidizer Vacuum

Conditions normalized using standard inlet pressures, temperatures, and densities.

PUMP EFFICIENCIES

Fuel, Isentropic

$$\eta_f = \frac{h_{OFIS} - h_{IF}}{h_{OF} - h_{IF}}$$

$$h_{OF} = f(P_{OF}, T_{OF})$$

Oxidizer, Isentropic

$$\eta_O = \eta_{OC}\,\gamma_O$$

$$\eta_{OC} = K_{40}\left(\frac{Q_{PO}}{N_O}\right)^2 + K_{50}\left(\frac{Q_{PO}}{N_O}\right) + K_{60}$$

$$\gamma_O = 1.000$$

$$K_{40} = -5.053 \qquad K_{50} = 3.861 \qquad K_{60} = 0.0733$$

TURBINES

Oxidizer, Efficiency

$$\eta_{TO} = \frac{B_{TO}}{B_{ITO}}$$

$$B_{TO} = K_5\,\frac{W_{PO}\;H_O}{\eta_O}$$

$$K_5 = 0.001818$$

$$W_{PO} = W_{OM} + W_{PUVO}$$

$$W_{PUVO} = \sqrt{\frac{Z_{PUVO}\;\rho_{OO}}{R_v}}$$

$$Z_{PUVO} = A + B\,(P_{OO})$$

$$A = -1597$$

$$B = 2.3828$$

$$\text{if } P_{OO} \geq 1010$$

$$\text{set } P_{OO} = 1010$$

$$\ln R_v = A + B\,(\theta_{PUVO}) + C(\theta_{PUVO})^3 + D\,(e)^{\frac{\theta_{PUVO}}{7}} + E\,\theta_{PUVO}\,(e)^{\frac{\theta_{PUVO}}{7}} + F\left[(e)^{\frac{\theta_{PUVO}}{7}}\right]^2$$

$$A = 5.566 \times 10^{-1}$$
$$B = 1.500 \times 10^{-2}$$
$$C = 7.941 \times 10^{-6}$$
$$D = 1.234$$
$$E = -7.255 \times 10^{-2}$$
$$F = 5.069 \times 10^{-2}$$
$$\theta_{PUVO} = 16.52$$

Fuel, Efficiency

$$\eta_{TF} = \frac{B_{TF}}{B_{ITF}}$$

$$B_{ITF} = K_{10}\ \Delta h_F\ W_T$$

$$\Delta h_F = h_{TIF} - h_{TEF}$$

$$B_{TF} = B_{PF} = K_5 \left(\frac{W_{PF}\ H_F}{\eta_F}\right)$$

$$W_{PF} = W_{FM}$$

$$K_{10} = 1.415$$

$$K_5 = 0.001818$$

Oxidizer, Developed Horsepower

$$B_{TO} = B_{PO} + K_{56}$$

$$B_{PO} = K_5 \left(\frac{W_{PO}\ H_O}{\eta_O}\right)$$

$$K_{56} = 0 \qquad K_5 = 0.001818$$

Fuel, Developed Horsepower

$$B_{TF} = B_{PF}$$

$$B_{PF} = K_5 \left(\frac{W_{PF}\ H_f}{\eta_f}\right)$$

$$W_{PF} = W_{FM}$$

Fuel, Weight Flow

$$W_{TF} = W_T$$

$$W_{TO} = W_T - W_B$$

$$W_B = \left[\frac{2K_7\ \ \gamma_2}{\gamma_2 - 1}\ (P_{RNC})^{\frac{2}{\gamma_{H2}}}\right]^{\frac{1}{2}} \left[1 - (P_{RNC})^{\frac{\gamma_{H_2} - 1}{\gamma_{H2}}}\right]^{\frac{1}{2}} \frac{A_{NB}\ P_{BNI}}{(R_{H_2} T_{BIR})^{\frac{1}{2}}}$$

$$P_{RNC} = f\ (\beta_{NB}, \gamma_{H_2})$$

$$\beta_{NB} = D_{NB}/D_B$$

$$\gamma_{H2},\ M_{H2} = f(T_{H2R}, R_G)$$

$$A_{NB} = K_{13}\ (D_{NB})^2$$

$$K_{13} = 0.7854$$

$$T_{BIR} = T_{T10} + 460$$

$$P_{BNI} = P_{TEFS}$$

P_{TEFS} = Iteration of

$$P_{TEF} = P_{TEFS}\left[1 - K_8\left(\frac{W_T}{P_{TEFS}}\right)^2 \frac{T_{H2R}}{D^4{}_{TEF}\, M_{H2}}\left(\frac{\gamma_{H_2} - 1}{\gamma_{H2}}\right)\right]^{\frac{\gamma_{H2}}{\gamma_{H2}-1}}$$

$$K_8 = 38.90$$

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11. SUPPLEMENTARY NOTES Available in DDC.	12. SPONSORING MILITARY ACTIVITY NASA, Marshall Space Flight Center (I-E-J), Huntsville, Alabama

13. ABSTRACT

Eight test periods involving a total of 14 starts of the J-2 rocket engine were conducted at pressure altitudes ranging from 93,000 to 111,000 ft. These tests were a continuation of an environmental verification and start transient investigation on a flight configuration J-2 engine (S/N J-2052). Firing durations ranged up to 50 sec; a total of 293.6 sec of engine operating time was accumulated during the test period. Unexpected excessive gas generator temperatures experienced on restart tests under simulated orbital restart conditions necessitated reorientation of the objectives for this test series. Satisfactory engine restart was subsequently obtained at turbine crossover duct conditions predicted for Saturn V (flight 501) restart with the propellant utilization valve in the full open position.

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*Huntsville, Alabama.

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rocket engines						
J-2 engines						
altitude testing						
performance						
ignition characteristics						
Saturn						

1. Rocket motors -- J-2
2. " " -- Performance
3. " " -- Ignition
4. Missiles -- Saturn.

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